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Payload Analysis for Space Shuttle Applications (Study 2.2) Final Report

Volume III: Payload System Operations Analysis (Task 2.2.1)

Prepared by
ADVANCED VEHICLE SYSTEMS DIRECTORATE
Systems Planning Division

15 October 1972

Prepared for OFFICE OF SPACE SCIENCE
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Washington, D. C.

Contract No. NASW-2301

Systems Engineering Operations
THE AEROSPACE CORPORATION

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TABLE OF CONTENTS

TABLE OF CONTENTS (cont.)

TABLE OF CONTENTS (cont.)

TABLE OF CONTENTS (cont.)

TABLE OF CONTENTS (cont.)

TABLES

TABLES (Cont'd)

TABLES (Cont'd)

TABLES (Cont'd)

FIGURES

FIGURES (Cont'd)

NOMENCLATURE

AGE	Aerospace Ground Equipment
ASO	Austere Solar Observatory
AVE	Aerospace Vehicle Equipment
CMG	Control Moment Gyro
CO	Checkout
CST	Combined System Test
EMC	Electro Magnetic Compatibility
EMI	Electro Magnetic Interference
I/F	Interface
IST	Integrated System Test
LSO	Large Solar Observatory
LUT	Launcher Umbilical Transporter
MMD	Mean Mission Duration
NPV	Net Present Value
OAS	Orbital Adjust Stage
OMS	Orbital Maneuvering System
P/L	Payload
PRD	Program Requirements Document
RF	Radio Frequency
RH	Relative Humidity
R&I	Receiving and Inspection
STDN	Spaceflight Tracking and Data Network
TDRS	Tracking and Data Relay Satellite
UTE	Unified Test Equipment

1. Introduction

1. INTRODUCTION

This volume describes the technical and cost analysis that was performed for the payload system operations analysis. The technical analysis consists of the operations for the payload/Shuttle and payload/Tug, and the spacecraft analysis which includes sortie, automated, and large observatory type payloads. The cost analysis includes the costing tradeoffs of the various payload design concepts and traffic models. The overall objectives of this effort were to identify payload design and operational concepts for the Shuttle which will result in low cost design, and to examine the low cost design concepts to identify applicable design guidelines (see Volume I).

The operations analysis examined several past and current NASA and DoD satellite programs to establish a Shuttle operations model. From this model the analysis examined the payload/Shuttle flow and determined facility concepts necessary for effective payload/Shuttle ground operations. The study of the payload/Tug operations was an examination of the various flight timelines for missions requiring the Tug.

The spacecraft analysis was a conceptual design effort for programs which would be representative of the Shuttle era. They consisted of sortie with a solar observatory payload as an example, communication satellites as examples of OA demonstration programs, and example observatory satellites. Contractor and agency study reports were obtained on these payloads to define the mission objective and equipment, and to describe the spacecraft system. The design effort was conceptual resulting in layouts and subsystem identification sufficient to estimate dimensional constraints, subsystem weights, and payload system cost differences. Conceptual layouts are consistent with the mission descriptions for the 1980s era (which are conceptual in all cases, i.e., drawings and specifications of instruments were not available). Furthermore, the Shuttle and Tug definition data were also of a descriptive nature.

The definition of the Shuttle and Tug as currently described in Ref. 1.1 through 1.4 were not available at the initiation of this study. It was therefore necessary to compile the best available Shuttle and Tug description, interface, and environmental data from all sources. This was documented as the mid-term report but was not published (Ref. 1.5). The data in the mid-term report is in substantial agreement with that given in Ref. 1.1 through 1.4, particularly in those areas which were important considerations in the payload design studies, and should not affect appreciably the completeness or results of the study presented in this volume. Users of these study results, however, should refer to the Shuttle references for payload accommodation information. The comparison of the various documented Shuttle descriptions is reported in Study 2.1 Final Report (Ref. 1.6).

The synthesis of an economical program concept with high scientific value requires a series of design/cost tradeoffs to determine lower cost configuration characteristics and operational modes. Potentials for low cost design and operations should continue to be thoroughly exploited in order to control the system costs, considering cost drivers such as development hardware quantities, spare satellites, flight hardware, scientific experiment requirements, satellite reliability, and expected number of Shuttle flights -- all in the context of operational modes available with the Shuttle. The cost analysis task addressed these objectives with an investigation of the three payload program concepts involving the use of the Shuttle in a sortie mode, automated payloads, and an observatory program. The designs for a variety of payloads were reviewed, program approaches were processed in a capture analysis, and cost estimates were prepared. Results of the cost exercise were compared in order to extract significant cost trends and tradeoffs. These comparisons were analyzed to identify lower cost program approaches for the example, show cost/value trends with scientific requirements, and develop programmatic guidelines. Low cost program approaches are identified and cost/scientific value tradeoffs are displayed.

REFERENCES

1.1 "Space Shuttle Program Requirements Document - Level NASA/OMSF, 21 April 1972, Revision No. 4

1.2 "Space Shuttle, Baseline Accommodations for Payloads," NASA/MSC, MSC-06900, 27 June 1972

1.3 Baseline Tug Definition Document, NASA/MSFC, 15 March 1972

1.4 Baseline Tug Definition Document, NASA/MSFC, Revision A, 26 June 1972

1.5 Payload Analysis for Space Shuttle Applications (Study 2.2), Mid-Term Report, Vol. I, Reusable Payload Design Guideline, 1 May 1972 (unpublished).

1.6 Space Shuttle Mission and Payload Capture Analysis (Study 2.1), ATR-73(7311)-1, The Aerospace Corporation, (31 August 1972) (Contract NASW-2301).

2. Payload Operations

2. PAYLOAD OPERATIONS

2.1 INTRODUCTION

Principal objectives of the payload operations analysis were to:

(1) Identify the differences between the conduct of payload operations with Shuttle technology and similar operations with existing expendable systems.

(2) Identify operational changes the Shuttle will produce that will be beneficial to NASA payloads in terms of performance and/or cost.

(3) Determine and establish the appropriate elements for costing purposes.

(4) Define payload design guidelines that will have an impact on the effective accomplishment of payload operations in the Shuttle era.

The approach involved reviewing historical records of the manner in which payload operations have previously been conducted with expendable systems, and studies and analyses conducted by various agencies with regard to the manner in which operations are likely to be conducted in the Shuttle era. These data were evaluated, and a synthesized flow of operations and timelines capturing the various approaches were generated to provide a study baseline for further examination. That baseline was refined by exercising the variables, the results were analyzed, and conclusions were developed. This approach resulted in considerable NASA/contractor documentation being reviewed with respect to payload operations and associated design, facilities, equipment, manpower, and costs.

2.2 PAYLOAD OPERATIONS FLOW AND TIMELINES

The key areas under consideration in the operations analysis task are illustrated as major milestones in Figure 2-1. The major thrust of this analysis

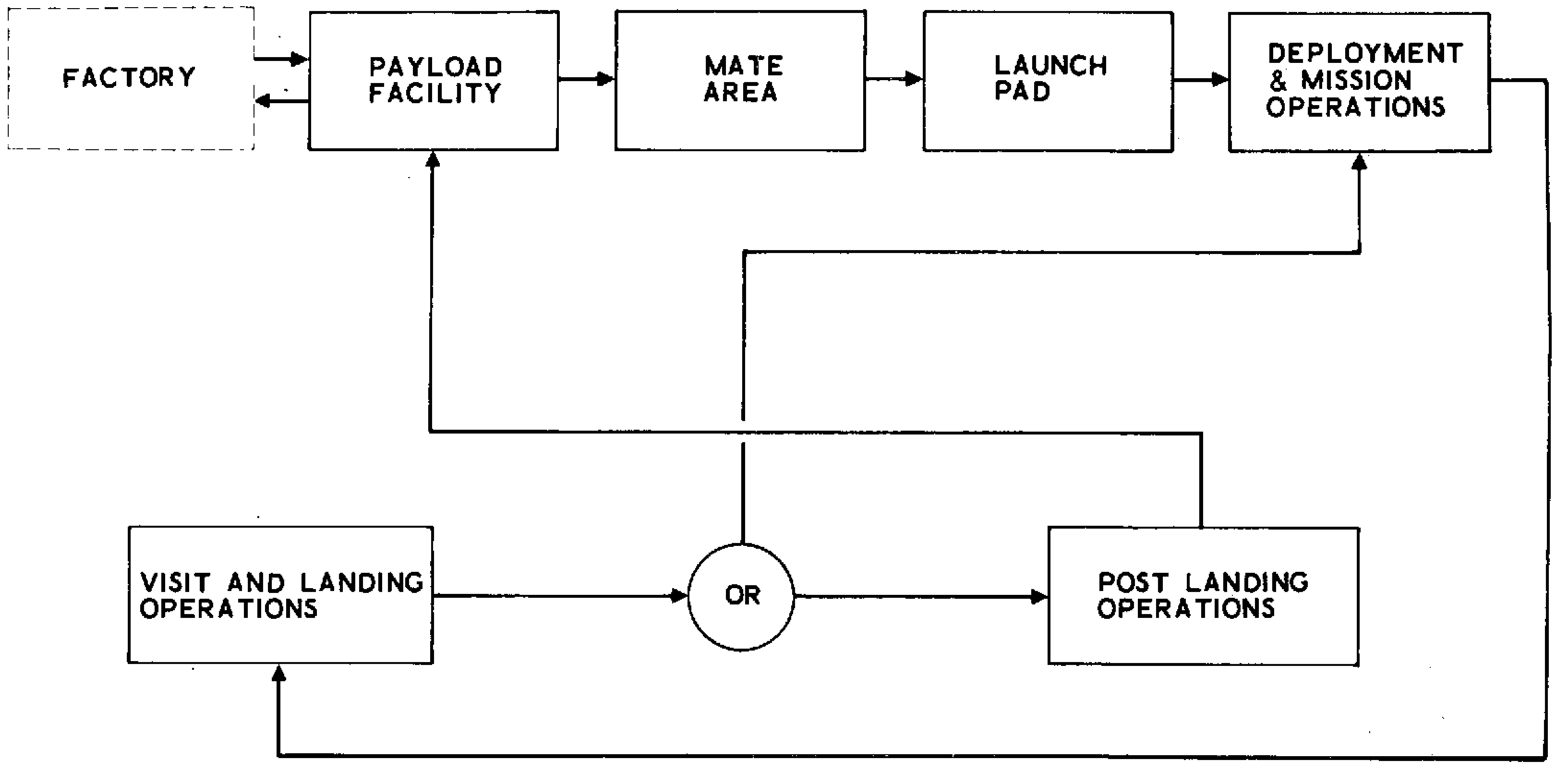


Figure 2-1. Operations Flow Analysis Major Milestones

effort has been directed toward operations conducted at the launch site from receipt of the payload through the pad activities. Payload operational procedures and problem-solving are centered in this activity area. It should be noted that on-orbit operations are, almost without exception, duplicated, tested, or otherwise carried forward on the ground, as well.

In the course of developing comparative operations, analysts studied detailed block flow diagrams of payloads including the Nimbus "D" in the expendable mode (Ref. 2.1); synchronous earth observatory, earth observation, and small research satellites in the Shuttle mode (Ref. 2.2); and the OAO/LST (Ref. 2.3) and two DoD satellites in both the expendable and Shuttle modes. A list of the payloads examined with launch site times is presented in Table 2-1. These payloads were selected because they were representative of type. A study baseline of operations was then generated, expanded into a major milestone operational flow, refined in detail, and exercised to establish that it captured the representative payloads. This detailed operations flow, presented in Figures 2-2 through 2-7, was based on Ref. 2.1, 2.3, 2.4, and detailed flow data from DoD programs.

Essentially the same documentation referred to above was reviewed as an approach to establishing a timeline for the payloads associated with the representative flow of operations. Examples of the timelines broken out for examination are presented in Figure 2-8 for Nimbus "D", in Figure 2-9 for two DoD payloads in the expendable mode, in Figure 2-10 for the low cost OAO and SEO in the Shuttle mode, and in Figure 2-11 for the HEAO in both the expendable and Shuttle modes. While many of the payloads examined had shorter timelines (see Table 2-1), the above examples are representative and cover the spectrum.

The review indicated that time for payload operations from receipt at the launch base through mating with the orbiter is roughly 20-25 days. It

Table 2-1. Program Timelines*

Program	Days in Simulator	Days Through Orbiter Mate
Astronomy Explorers	7	31
OSO	4	19
Relativity	2	14
Radio Interferometer	3	15
HEAO	4	25
Large Space Telescope	6	32
Large Solar Observatory	4	28
Large Radio Observatory	4	22
Intermediate Astronomy Instruments	3	16
Aeronomy & S/E Sortie	3	12
IR Astronomy	3	18
Physics Lab Sortie	4	14
Space Station (Phys. Lab)	1	17
Cosmic Ray Lab	3	19
Life Sciences Sortie (Zoo)	4	21
Life Sciences Sortie (Bio)	4	21
Life Science Lab	3	24
Applications Tech. Sat	3	13
Medical Network	4	11
Comm. & Nav. Sortie	2	14
Comm. & Nav. Research	4	17
Polar Earth Obs.	4	26
Sync. Earth Obs.	4	20
Earth Physics Sat.	6	28
Tiros Class	5	16
Polar Earth Resource	4	19
Earth Resources Sortie	3	19
Earth Obs. Lab	2	22
Material Sciences	3	8
Tech. Sortie	2	22
Space Mfg. Station	4	16
Crew Cargo Module	5	31
Gen. Purp. Lab. Mod.	2	14
Orbiter - P/L	6	25
Orbiter - Pallet - P/L	6	26
Orbiter MSM - P/L	6	25
Orbiter - Tug - P/L	7	32
Orbiter-Tug-Kick-P/L	8	37
DoD	5	21
Viking	5	25
Venus Explorers	4	27
Jupiter Pioneer Orbiter	4	19
Asteroid Survey	2	11
Grand Tour	11	49
Mars Sample Return	7	60

Average Days in Simulator - 4 (1-8) ------------ 8 out of 43 Types over 5 Days
Average Days thru Orbiter Mate - 21 (11-37) ---- 8 out of 43 Types over 25 Days
4 out of 43 Types over 30 Days

*Ref. 2.6 and 2.7

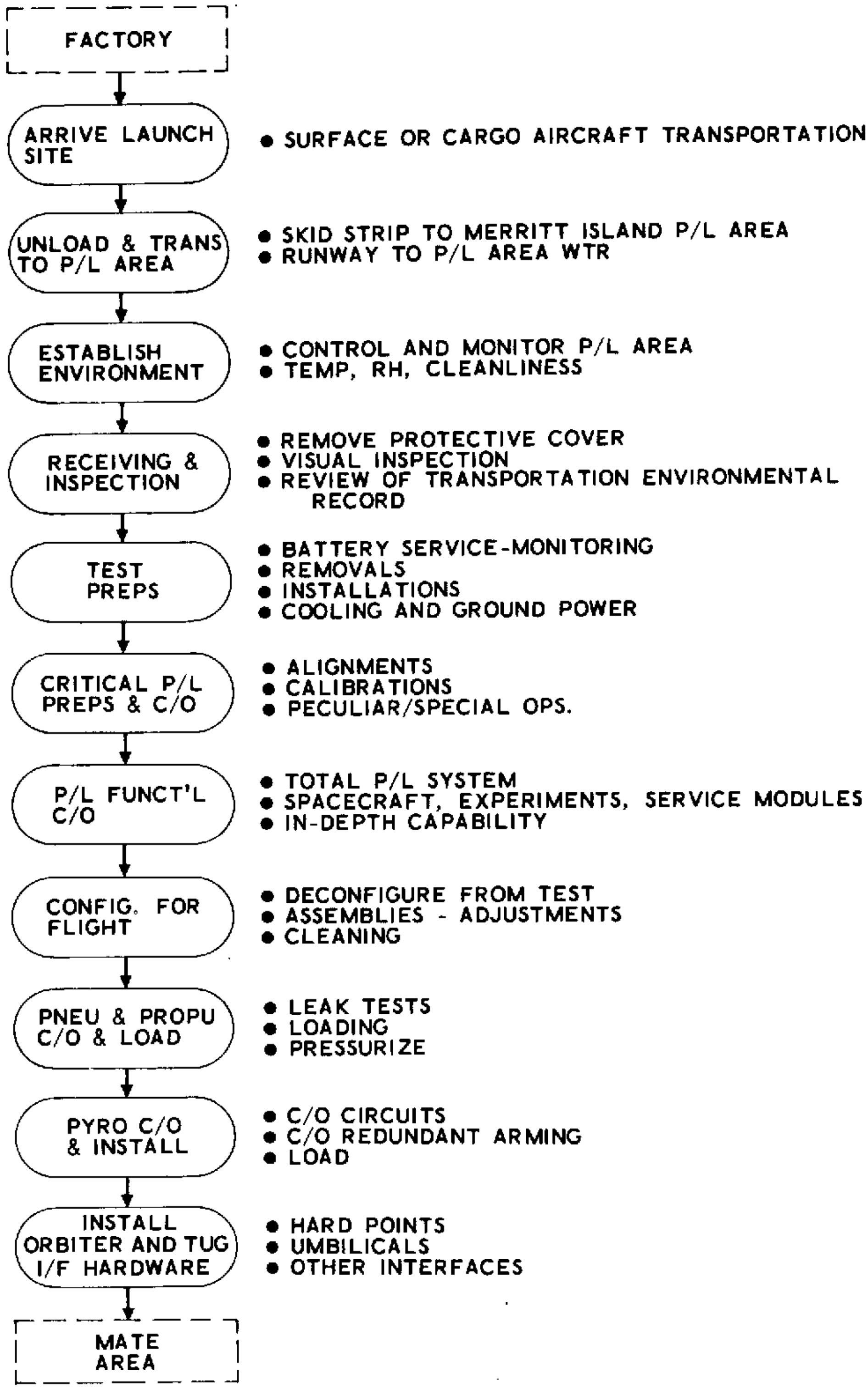


Figure 2-2. Payload Facility Flow of Operations

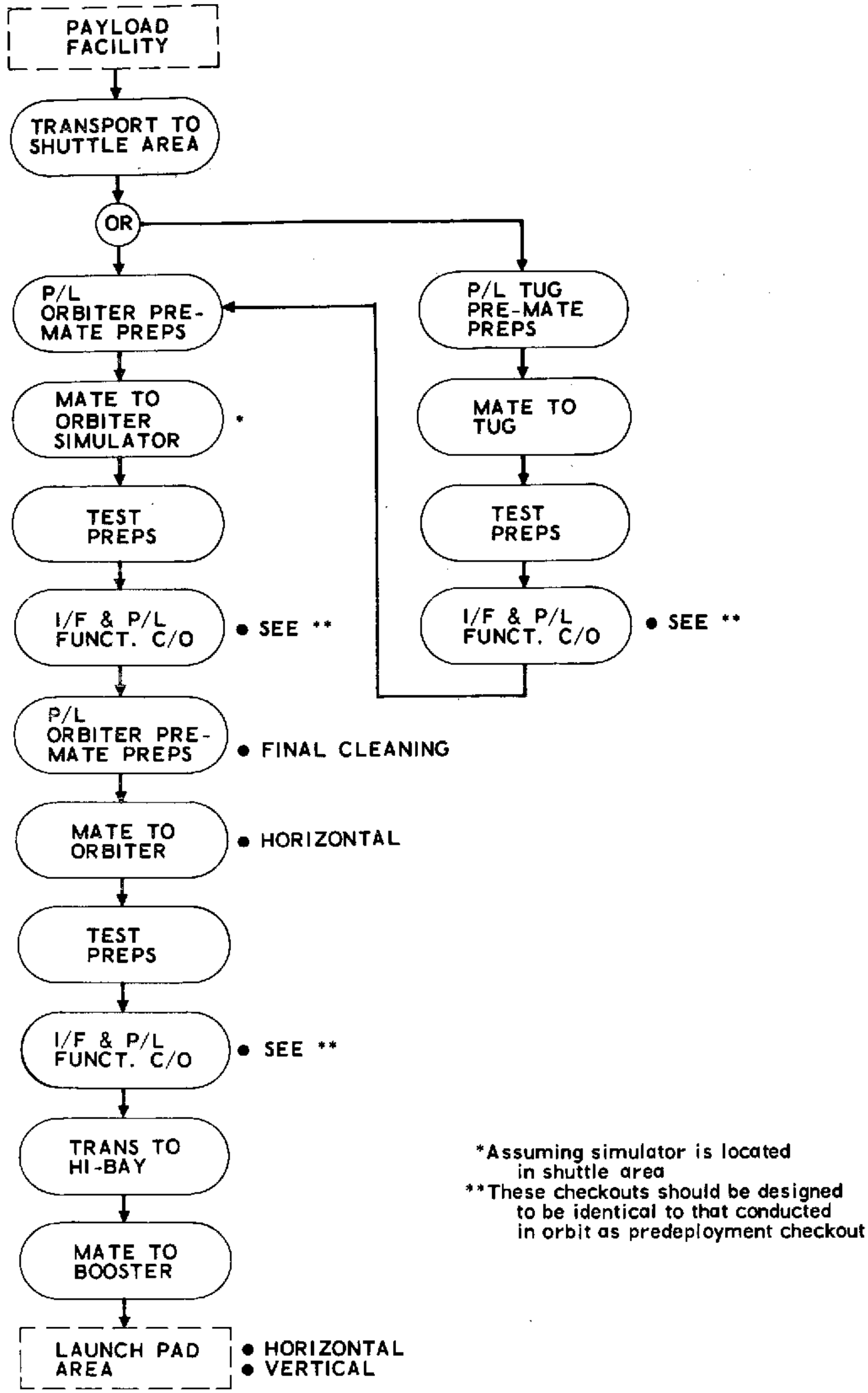


Figure 2-3. Mate Area Flow of Operations

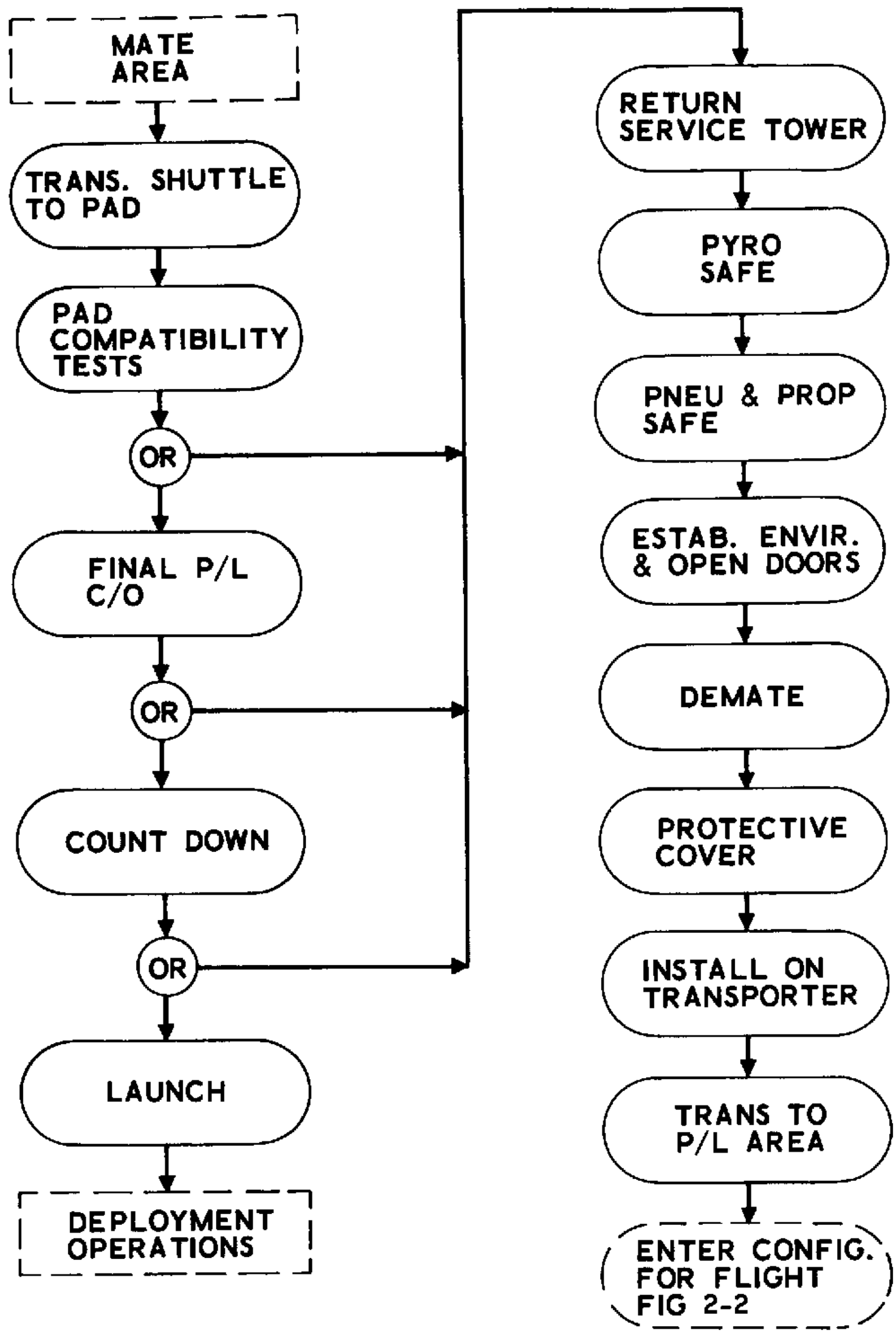


Figure 2-4. Launch Pad Flow of Operations

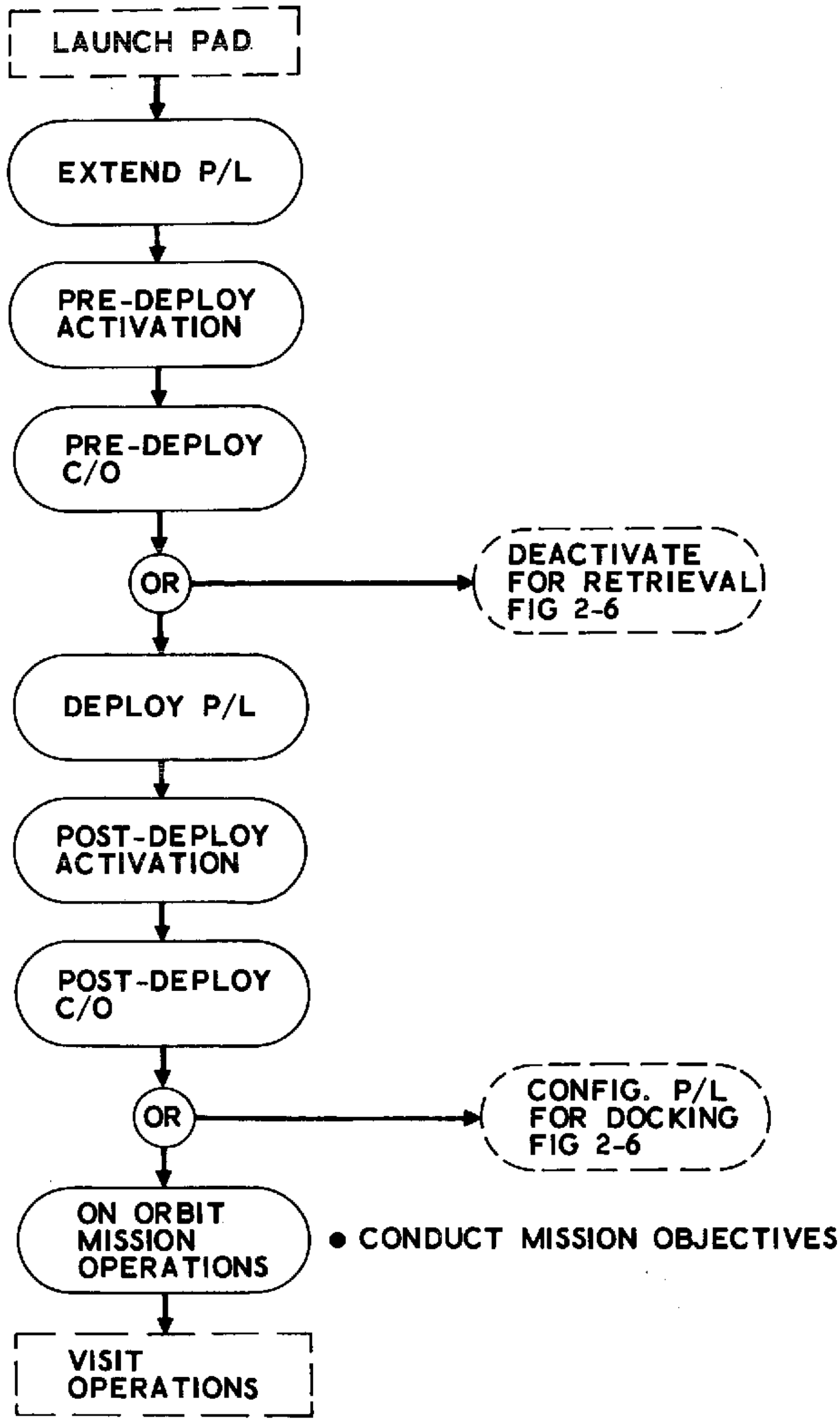


Figure 2-5. Deployment and Mission Flow of Operations

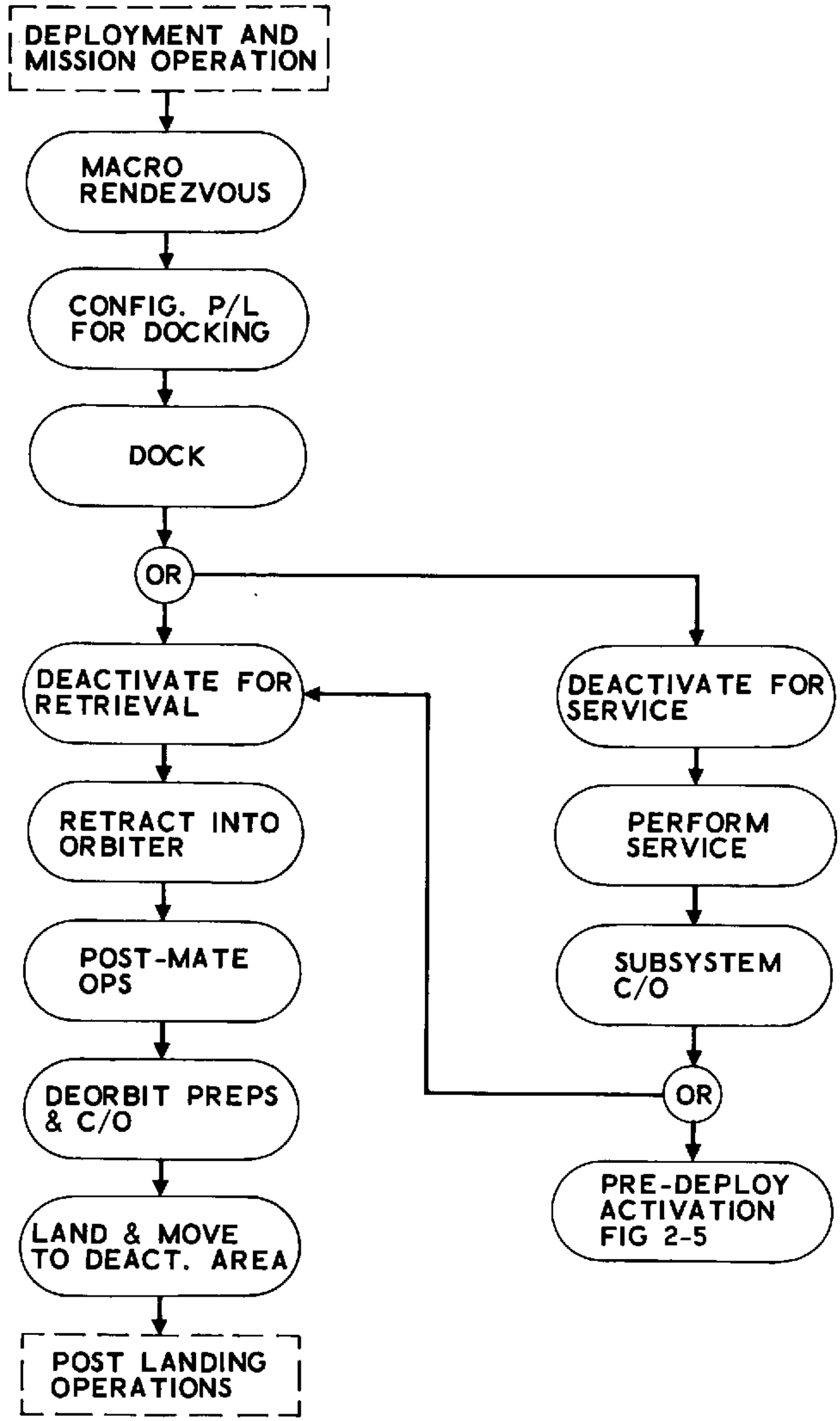


Figure 2-6. Visit and Landing Flow of Operations

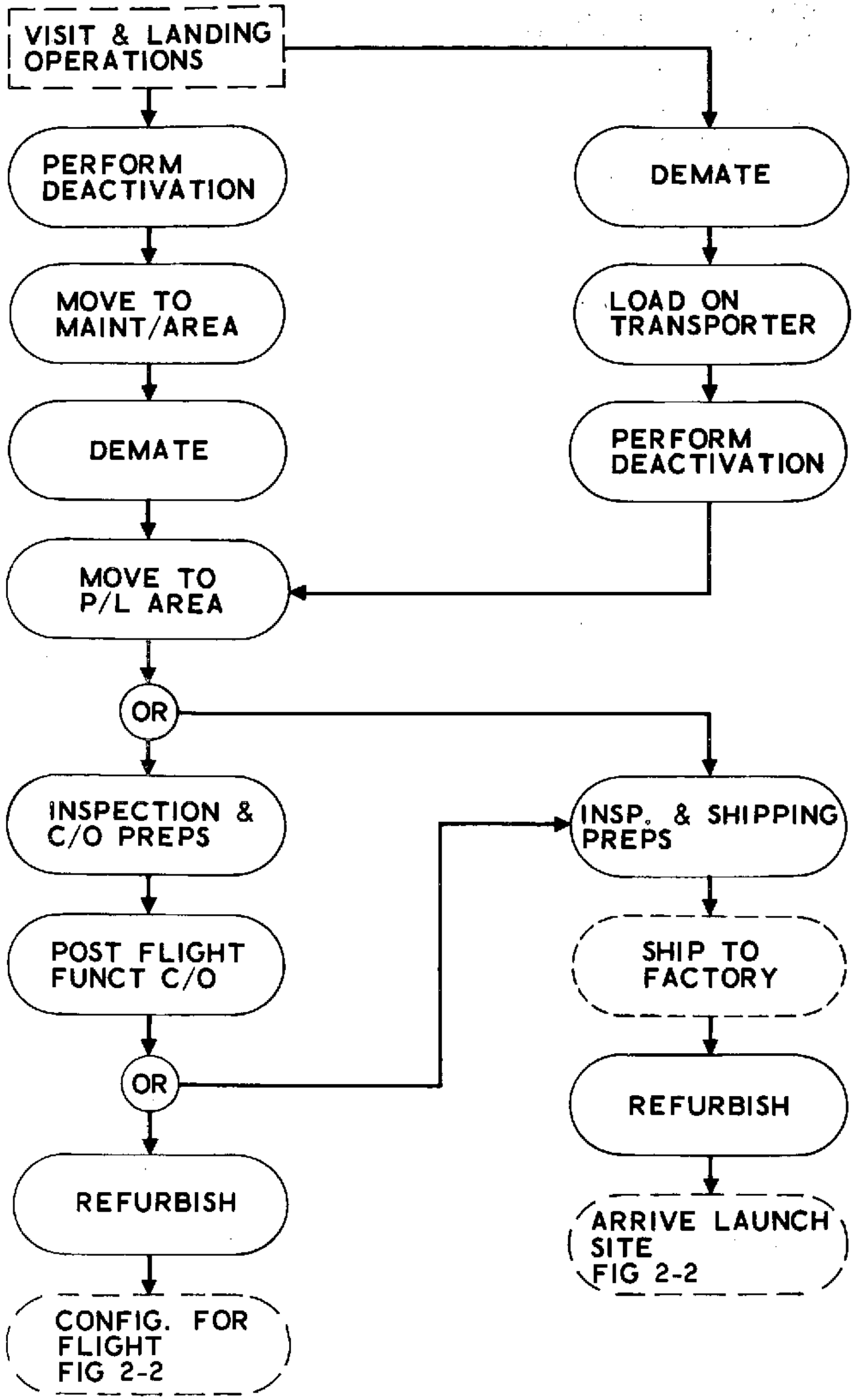


Figure 2-7. Post Landing Flow of Operations

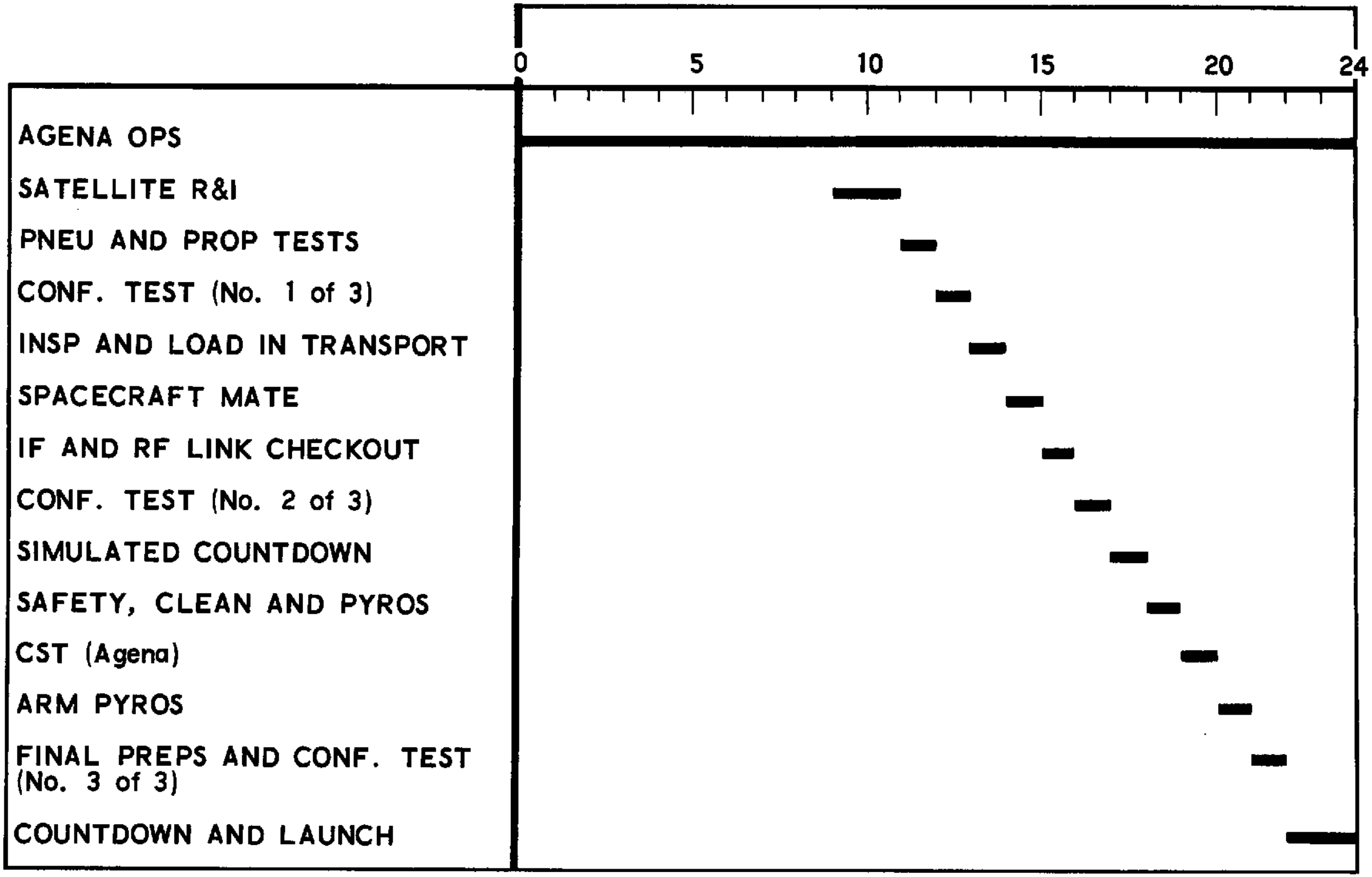


Figure 2-8. Timelines - Nimbus D Thorad/Agena Launch

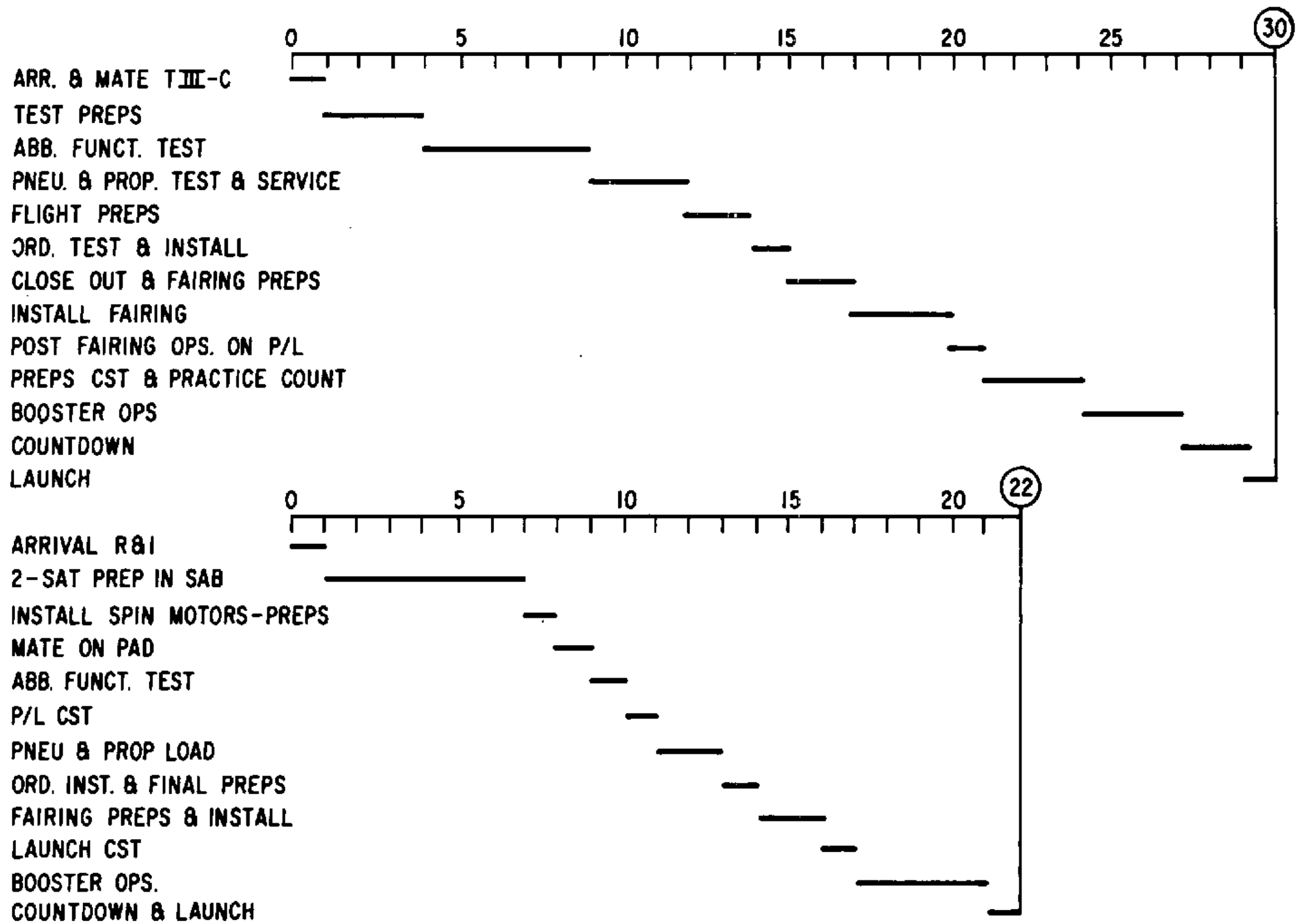


Figure 2-9. Timelines - DoD Number Programs, T III-C Launch

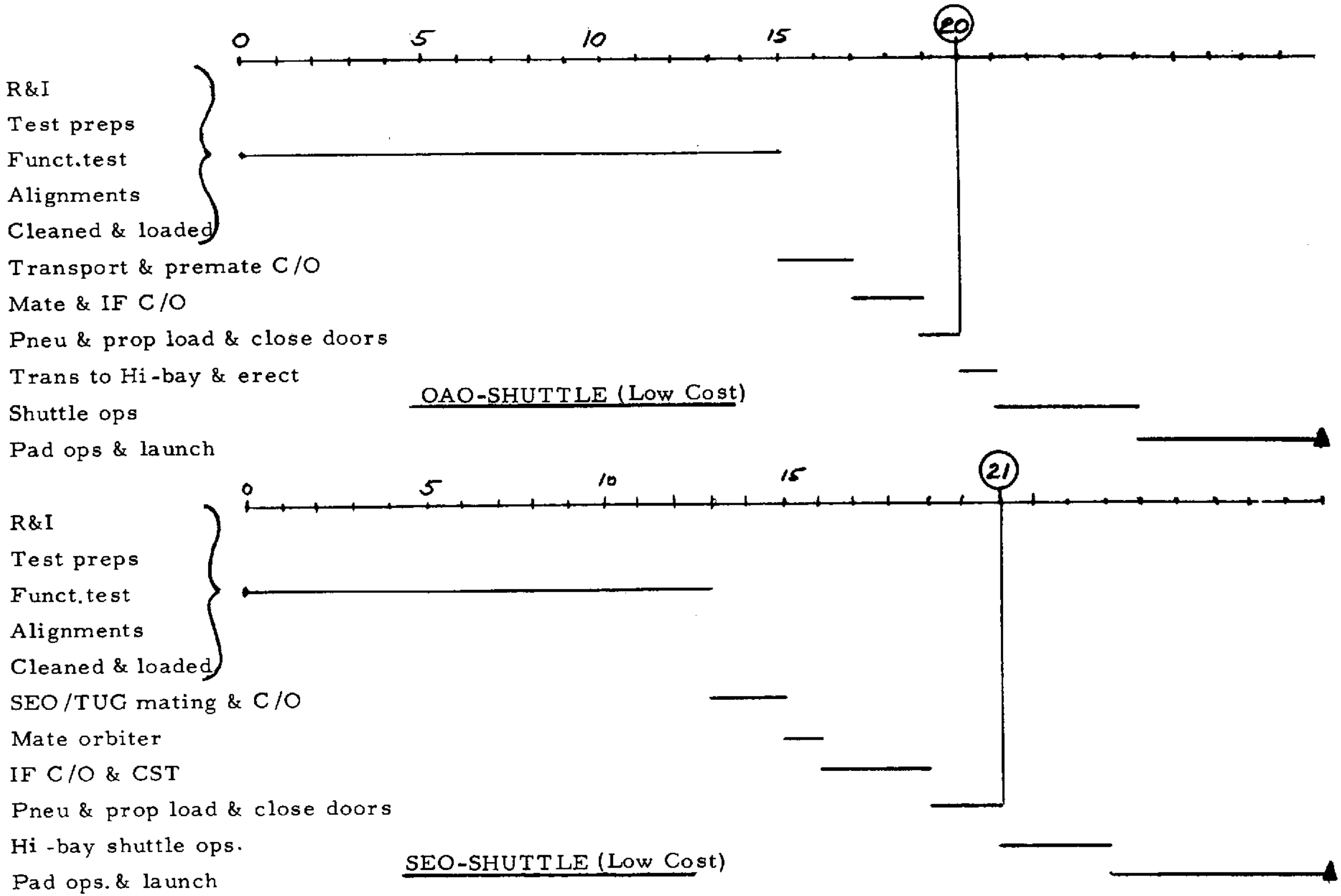


Figure 2-10. Timelines - OAO and SEO, Shuttle Mode (Ref. 3.8)

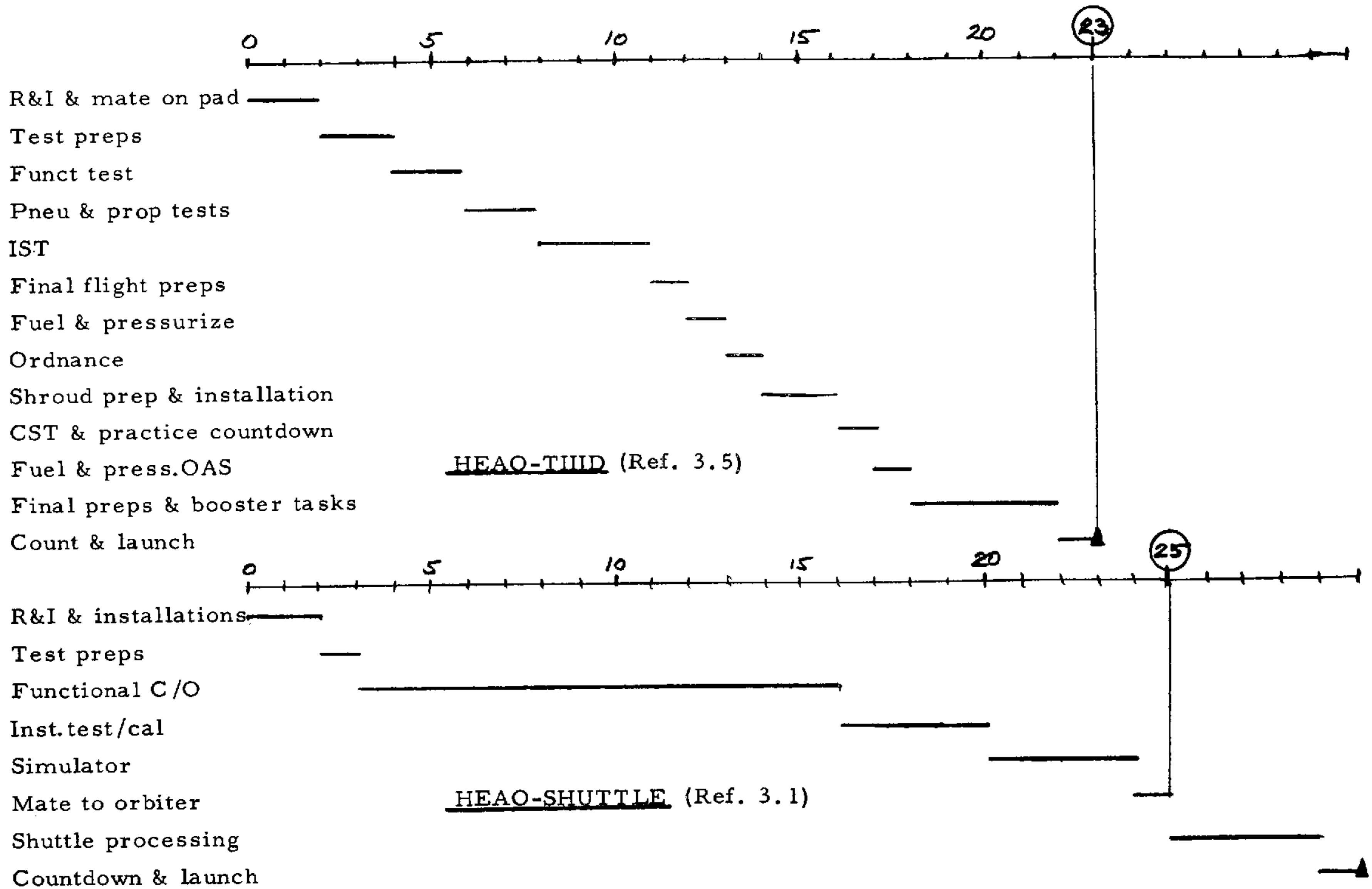


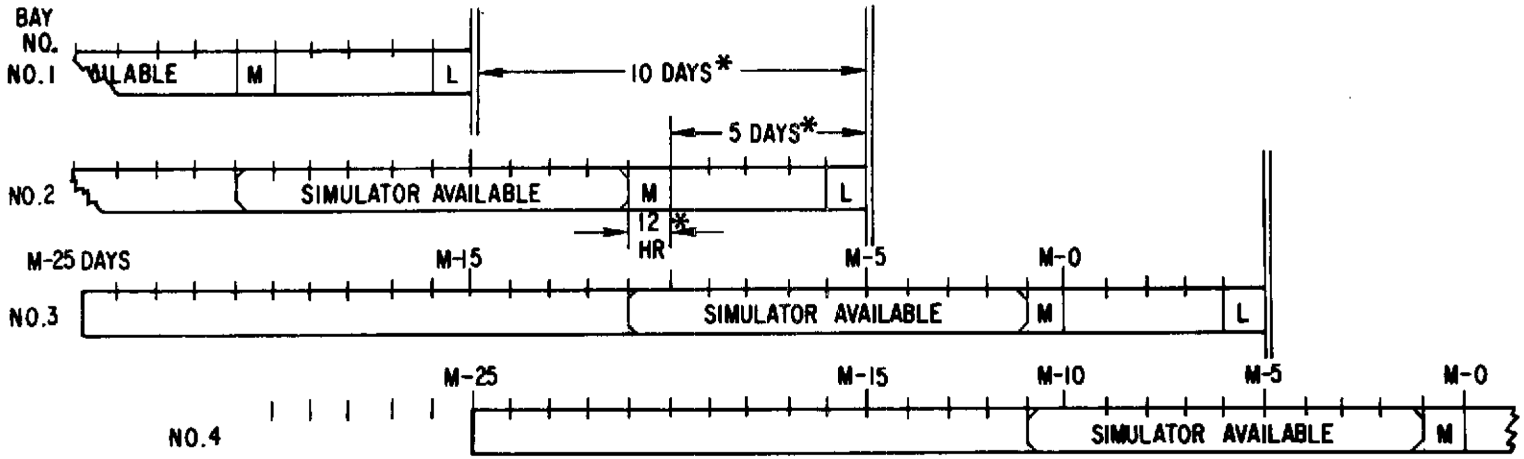
Figure 2-11. Timelines - HEAO, Expendable and Shuttle Modes

should be noted that the use of an orbiter simulator for payload/orbiter pre-mate integration and testing purposes is not included in the timeli except for the HEAO in the Shuttle mode. This example illustrates th effect of using an orbiter simulator in the flow of operations. While t timeline is slightly extended over the other examples, the payload can be mated, interfaces checked out, tests performed, and any problem areas identified and resolved during the time the orbiter is in the simulator. Identical tasks may be expected to be accomplished without problems when the payload and the orbiter are actually mated. This simulation reduces the time in the orbiter flow of operations and ultimately preserves the integrity of the planned launch schedule, both of which considerations, of course, are important in the Shuttle mode. Both considerations, incidentally, are absent in the other examples. Consideration of the foregoing was sufficient to cause 25 days to be selected as a basic time required for the payload pre-launch operations flow.

2.3 SYNTHESIZED TIMELINE

The synthesized timeline shown in Figure 2-12 was generated on the basis of the 25-day payload flow, inputs from contractor Shuttle studies, and some assumptions. The figure is generally self-explanatory; some further explanation, however, is in order.

The values indicated as Shuttle baseline inputs have been used in many studies and are not fixed. They are credible, however, and since they are treated as variables they are satisfactory for use as a baseline. The 10-day launch center provides a launch rate sufficient to handle early system traffic. The five days indicated as required for Shuttle operations, following mating of the payload on M day at L-5 days (Ref. 2.5), is a maximum. Some alternate plans indicate this activity at L-1. The time made available for payload mating has ranged from as little as four hours to as much as 36 hours.



* SHUTTLE BASE LINE INPUTS:

- 10 DAYS BETWEEN LAUNCH CENTERS
- 5 DAYS FOR SHUTTLE OPS. INCLUDING COUNT-DOWN AND LAUNCH
- 12 HRS. AVAILABLE FOR MATING P/L TO ORBITER ON M-DAY

- 3 PAYLOADS IN PROCESS SIMULTANEOUSLY IN PAYLOAD FACILITY
- SIMULATOR IS AVAILABLE FOR USE UP TO 10 DAYS FOR EACH PAYLOAD
- 25 DAY NORMAL PAYLOAD FLOW (ARRIVAL TO ORBITER DOORS CLOSED ON M-DAY)
- 30 PLUS DAYS CAN BE HANDLED
  - #4 P/L COULD ENTER FACILITY BAY AS EARLY AS #1 GOES IN SIMULATOR
  - RISK OF #1 RETURNING TO BAY DUE TO PROBLEMS OR NO LAUNCH
- EFFECT OF MOVING M-DAY FROM L-5 DAYS TO L-1 DAY:
  - REDUCES STANDBY MANPOWER 4 DAYS FOR EACH PROGRAM
  - INCREASES PAYLOAD RELIABILITY (SHORTER C/O TO LAUNCH TIME)
  - DECREASES TIME OF RECOGNITION OF LAUNCH SLIP OR CANCELLATION
  - ENHANCES SECURITY FOR DOD (EXPOSURE TIME)
  - REQUIRES VERTICAL MATE OF PAYLOAD TO ORBITER
  - INCREASES SHUTTLE AVAILABILITY FOR PRIORITY PAYLOAD
  - REDUCES STANDBY PAYLOAD BAY TIME FOR EACH PROGRAM

Figure 2-12. Synthesized Timeline, Payload Pre-Flight

Twelve hours is a minimum requirement. The first three bars in the figure are the timelines for three payloads in their respective bays in the payload facility. The last bar represents the flow of the #4 payload arriving at the launch base to enter the payload bay vacated by the #1 payload. Some of the flexibility of the system in handling a mix of payloads is revealed in that the time available for a payload to use the orbiter simulator is normally 10 days. The actual time the simulator is used, as well as how much of the available 25 days is used, is left to the discretion of the payload program. For instance, #3 payload could have been scheduled to arrive and to enter the flow at M-10 days, to use the simulator for three days including a day of slack time, to mate, and to meet the 10-day launch center schedule without affecting the other two payload flows. Other schedule flexibilities are functions of the variables existing in the system.

2.4 OPERATIONS TRADEOFF ANALYSIS

The operational variables shown below were investigated by means of the synthesized timeline serving as a base for further analyses. There is considerable interdependency in the variables, so an attempt has been made to parameterize each of them so effects may be observed as a result of changing any one or a combination of the variables as inputs are received or generated.

(1) Shuttle launch centers - approximately 8 through 25 days

(2) Work week - 5 days vs 7 days

(3) Payload processing bays and time - 1 vs 2 or 3 in payload facility

(4) Payload-to-orbiter mate day - L-5 days vs L-1 day

(5) Payload/Shuttle C/O - simulator vs orbiter

The relationship between launch frequencies and launch centers was investigated. Data concerning that investigation are shown in Figure 2-13. The launch center variation of from 8 to 25 days can vary the launch frequency

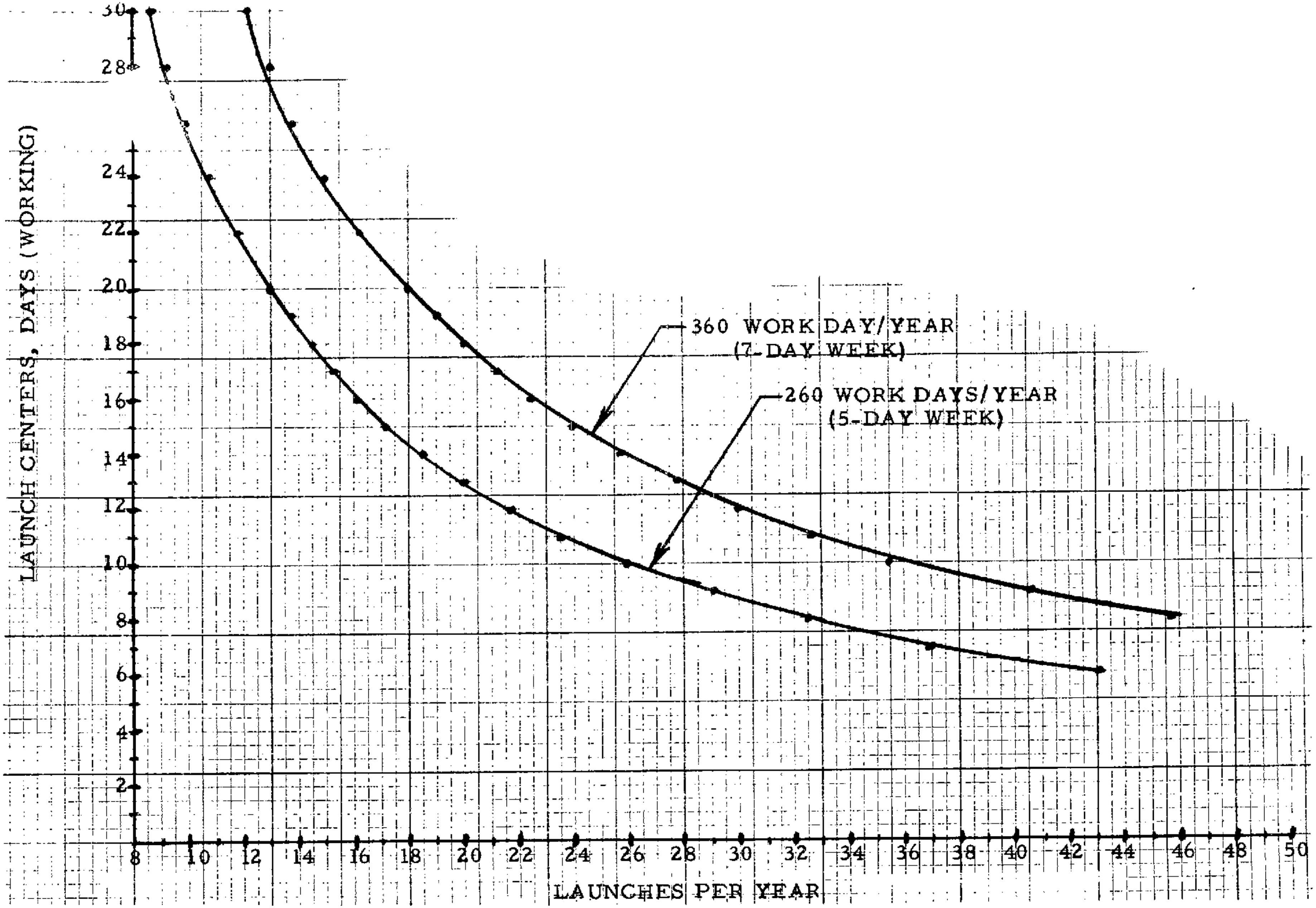


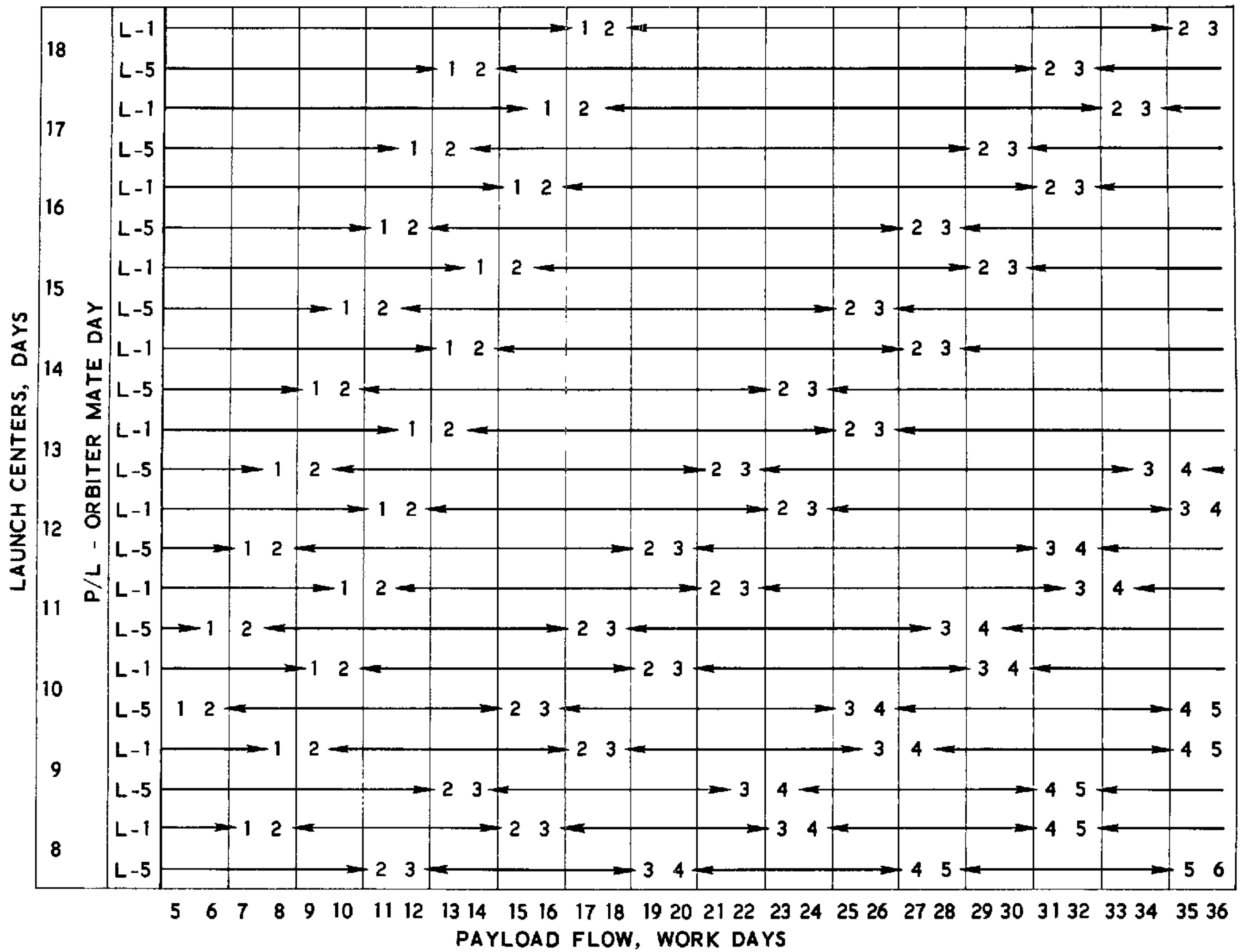
Figure 2-13. Launch Frequencies

from 10 to 32 launches per year. Examination of several typical Shuttle launch traffic models revealed that approximately 20 launches per year per launch site could occur in the early phases of Shuttle era. The 20 launches per year would result in 13-day launch centers. The 10-day launch center selected in the synthesized timeline appears to be justified in that it could accommodate up to 26 launches per year.

The comparison of 5-day vs 7-day work weeks is also shown in Figure 2-13. The 7-day work week vs the 5-day work week would provide nine more launches per year and for 10-day launch centers. This would increase the launch rate by 35 percent. Similarly, the 7-day work week could increase the launch center from ten to 14 days for 26 launches per year. The 7-day work week could also be used to catch up schedules in the event of payload delays and other slippages.

The number of payload processing bays required in the payload facility is a function of the average number of days in the payload flow of operations, the intervals between launch centers, and when the payload is mated to the orbiter. These variables may be exercised by using Figure 2-14, which reveals that by using the 13-day launch center for 20 launches per year, and by selecting the payload-to-orbiter mate day to occur one day before launch (L-1), only one payload processing bay will be required for payloads having flow times under 12 days. A mix of payloads having flows ranging from 12 through as many as 25 days can be handled by providing two processing bays. For a 10-day launch center, three processing bays will accommodate from 16 to 25 days in the processing bays for L-5 mate day. These examples indicate that three processing bays for 20 launches per year are needed to provide some time margin in the processing bays.

The advantage of mating on L-1 day as compared to L-5 day can be seen in Figure 2-14. The L-1 day basically provides for four more days in the



- ARRIVAL THROUGH MATE-TO-ORBITER
- BAY DEDICATED TO P/L THROUGH LAUNCH

Figure 2-14. P/L Processing Bays Required in P/L Facility processing bay. These four additional days can be used to shorten the launch center to eight days and correspondingly increase the launch frequency to 33 launches per year, or to decrease the number of payload processing bays from three to two if the payloads can be processed within 19 working days. It is evident then, that mating of the payload to the orbiter should be accomplished as close to launch day as possible with L-1 day preferred. From these timelines it can be observed that some type of orbiter simulator is required if these schedules are to be maintained.

These timelines are based on the orbiter flow time constraints imposed, where time allocations for payload mating are minimal and subsequent time to meet firm launch center schedule commitments does not allow time for problems. There are other factors that would also appear to support the need for an orbiter simulator including preventing potential damage to the orbiter or the payload, providing early knowledge of launch cancellation due to payload or other problems, providing mission-peculiar "training" capability for flight crews, and improving the ready stand-by status of priority payloads.

2.5 REDUCED PAYLOAD FLOW OF OPERATIONS

The potential for reducing the payload flow of operations and timelines was investigated for a typical autonomous payload, which is one defined as having a minimum or clean interface with respect to the orbiter. The following assumptions and ground rules were established in this investigation:

(1) The payload is complete and "flight ready" when received at the launch base. Small non-critical items are excepted including protective covers, delicate external items such as sunshades, special supports for transportation, and other accessory type items.

(2) The payload is optimized with respect to standardization of spacecraft or subsystems.

(3) The payload is configured for modular subsystem replacement for maintenance and refurbishment, and compatible spares are shipped with the payload per the low cost concept (Ref. 2.2).

(4) The payload is man-safe. This includes fluid, propulsion, pyro, and all other systems.

(5) The payload is designed to use "standardized" or common test, checkout servicing, and other support equipment furnished at the launch base for payload use. This includes unified test equipment (UTE) or equivalent universal type equipment.

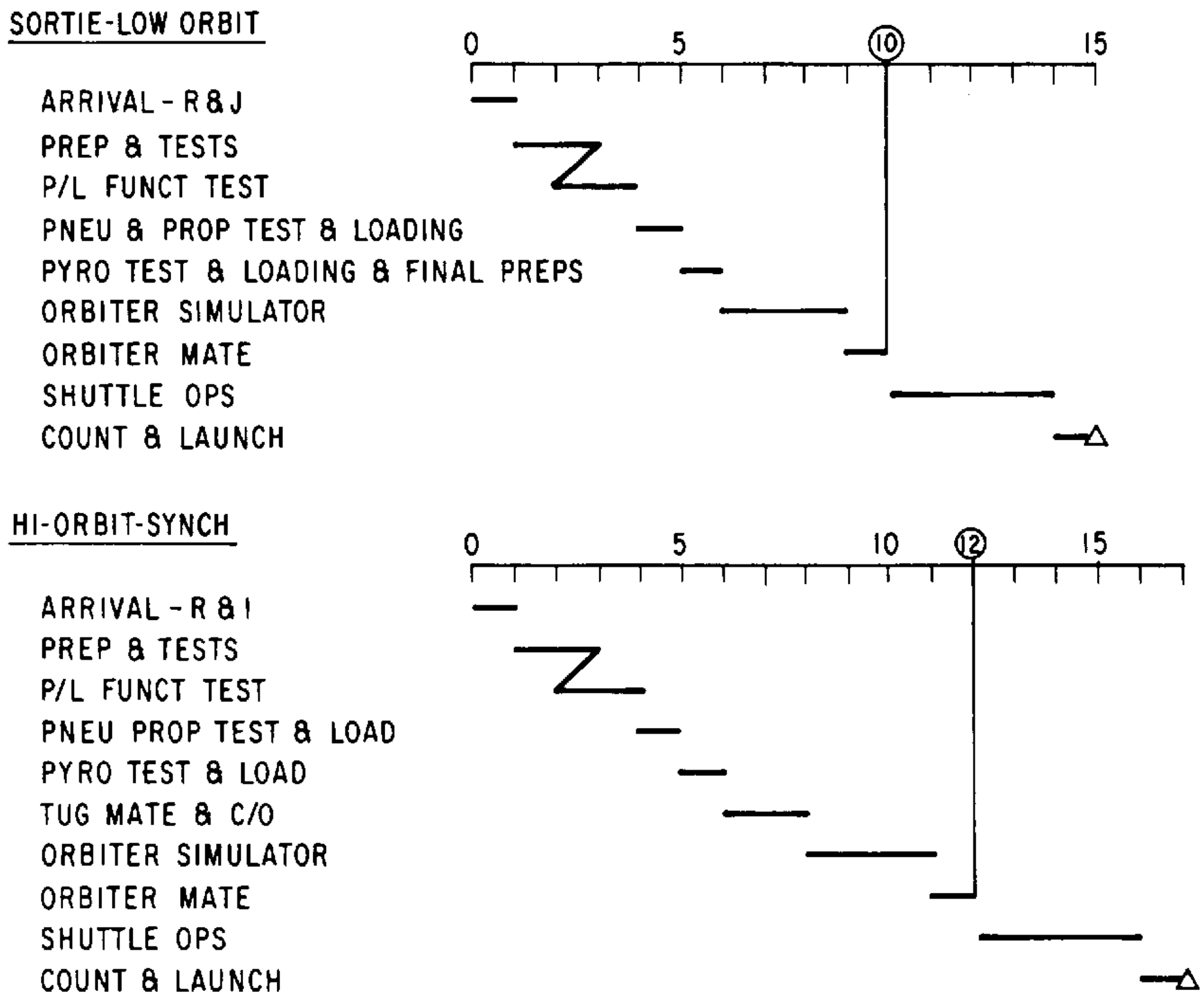
On the basis of these conditions, the following description of the flow of operations for a typical automonous payload in low, high, or synchronous orbit was generated and time requirements were estimated. The flow timelines are illustrated in Figure 2-15. It will be noted that it appears to be feasible and reasonable to expect that flows and timelines may be reduced to approximately half of that presently envisioned. The flow timelines establish the following conditions:

(1) Arrival, receive, and inspection (1 work day)
   - (a) Unload and position in payload processing bay
   - (b) Establish environment in bay
   - (c) Evaluate transportation environment tape
   - (d) Remove from transporter, set up workstands
   - (e) Inspect payload and store accompanying equipment

(2) Test preparation (1.5 work days)
   - (a) Minor removals and installations
   - (b) Critical alignments, calibrations
   - (c) Start test preps

(3) Test (1.5 work days)
   - (a) Finish test preps
   - (b) Conduct payload functional checkout
   - (c) De-configure from test

(4) Pneu-propulsion (1 work day)
   - (a) Leak test - load and pressurize
   - (b) Start flight configuration preps



*1. PAYLOAD ARRIVES AT LAUNCH BASE READY FOR LAUNCH   2. PAYLOAD DESIGNED FOR CLEAN INTERFACE

Figure 2-15. Approach to Autonomous Payload-Clean Interface

(5) Pyro, or other deployment devices (1 work day)

   (a) System checkout and load
   (b) Finish flight configuration preps
   (c) Preps for mating and moving

(6) Tug/Pallet/Modules, if utilized in mission (2 work days)

   (a) Move, mate, and check out all interfaces
   (b) CST, including payload functional test

(7) Orbiter Simulator (3 work days)

   (a) Mate and check out all interfaces
   (b) CST, including payload functional checkout
   (c) Prep for move to Shuttle area

(8) Orbiter (1 work day)

   (a) Transport to orbiter area
   (b) Mate and check out all interfaces
   (c) CST, including payload functional checkout

This flow and timeline results in 12 working days from receiving to orbiter mate and 10 working days from receiving to orbiter mate if Tug/Pallet/ Modules are not utilized.

2.6 STANDARDIZED GROUND EQUIPMENT AND FACILITIES

The standardized or common support equipment to be provided at the launch base for use by the payloads is important to the attainment of reduced payload operations. This equipment must be utilized by all payloads designed for the interfaces. Such a stipulation will preclude the requirement that each payload provide its own peculiar equipment, which is the practice today. This should result in lower costs as related to each program as well as the total mission model over the years. The basic criteria for the equipment follow.

(1) Provide standardized design that is common to, and will support requirements for, all mission model payloads to the maximum cost effective extent.

(2) Incorporate modular design to encompass the ranges of payload requirements. This applies to mechanical as well as electronic equipment and provides a means to accommodate such variables as fluid flow, pressures, power, signal input/output limits, payload size, and center of gravity locations. It also facilitates maintenance and increases the availability of the AGE.

(3) Design for maximum utilization at all points in the flow of operations. This includes the prelaunch processing and orbiter mating areas, launch pad, post-flight safing, refurbishment and storage areas, and alternate landing sites.

(4) Design to reduce equipment quantities by providing equipment sharing capability. Incorporate mobility, electronic transmission, and other means to permit equipment to be quickly shifted between payloads and locations.

(5) Design equipment for expansion capability to accommodate potential growth factors in payload requirements and numbers.

The problem of implementing the concept of providing standardized or common support equipment at the launch base is neither difficult nor unprecedented in function. The equipment should be furnished, controlled, and maintained by a launch base support agency. This agency should provide a Shuttle Users Payload Support Handbook to payload programs. Such a document would provide information and technical data pertinent to the equipment, facilities, and services available at the launch base and should be used by the program contractor to accomplish compatible payload design and planning. The payload program should utilize a Program Requirements Document (PRD) to request the support required, indicate schedules, and generally facilitate planning by the launch base agency.

An effort was made to define the equipment requirements. An approach was made by generating a list indicating the areas in the flow of operations where it would be used, and defining the quantities estimated to be required. (See Table 2-2). The indicated equipment quantities are based on a postulated

Table 2-2. Launch Base Payload Ground Support Equipment

Payload Ground Support Equipment		Payload Operations use Areas*						Quantities		
Equipment/Facility	Activity	Process Bays(3)	Orbiter Mate(1)	Launch Pad(1)	Post Flight(1)	Refurbish Bays(2)	Storage Bays(2)	Concurrent in use	Spares	Total
Unified Test Equipment	computerized universal system - inter-facility capability	X	X	X		X	X	TO BE DETERMINED		
Ground Power 115/400 VAC, 28 VDC	supply, control, monitor	3	1	1	1	1		7	2	9
Battery	activate, charge, monitor, exercise, test	3	1	1		1	1	7	2	9
Propulsion - Fuel/oxidizers	accurate load, purge, flush, vent, test	1		1	1			3	1	4
Pneumatics - liquids and gases	load, purge, flush, vent, test	2		1	1	1		5	1	6
Hydraulics	load, purge, flush, test	1				1		2	0	2
Leak Test - Tent/Booth	tracer gas measuring & isolation capability	1				1		2	1	2
Vacuum - high and low source	for operatating experiments, etc.	1				1		2	1	3
Pyrotechnic	item and system checkout	1	X	1	1	1		4	1	5
Alignments	structural and optical	1				1		2	1	3
EMI/EMC	generators and measuring capability	1	X	X		1		2	1	3
Cleaning	vacuum, flushing, measuring, etc. system level	2	X			1		3	1	4
Solar Power Set	simple checkout - panel isolation capability	1				X		1	1	2
Telemetry	coax, slave antenna - RF and hardwire links	1	1	1		1		4	1	5
Communications	RF generator, standards and measuring	X	1	X		1		3	1	4
Cooling Air - Bay and Umbilicals	controlled environment, monitor	1	1	1	1	1		5	1	6
Cryogenic Cooling	supply, control and monitor	1	X	1	X	X		2	1	3
Coolant	glycol type supply, control, monitor	1	X		X	1		3	1	4
Orbiter Simulator	form and fit, perform interface and P/L status checkout	1						1	0	1
Satellite Fixture	orbiter fittings - adj. height, rotates, mobile	3	X	X	X	X	2	7	1	8
Module fixture	orbiter fittings - adj. height, rotates, mobile	1				1	1	3	1	4
Pallet fixture	orbiter fittings - adj. height, rotates, mobile	1				1	1	3	1	4
Upper Stage fixture	orbiter fittings - adj. height, rotates, mobile	1						1	0	1
Access Stands	modular, adjustable, portable	3	1	1	1	2	1	9	1	10
Payload Transporter	payload totally assembled, environment controlled	1	X	X	1			2	1	3
Handling Group	slings, dollies, fork lifts, prime movers	3	X	X	X	1	X	4	1	5
Vapor Detection	propellant safety monitoring	3	1	1	1	1	1	8	1	9

*based on postulated operating system with 20 launch per year capability, equipment sharing is incorporated

X - equipment shared on as-required basis operating system in which, for instance, three payload processing bays and one launch pad are utilized. It should also be noted that a degree of equipment sharing was injected. This is evident where use in a particular area is indicated, although the numerical quantity is omitted. Requirements for upper stages such as the Tug, Agena, and Centaur are not included in the listing. Requirements outside those listed will be considered as program-peculiar and will be the property of, and furnished by, the affected program agency. This equipment should satisfy requirements for the initial activation of the Shuttle payload operations at either VAFB or KSC and could be expected to support early operations up to approximately 20 launches per year. Increases in any of the operations use-areas, for any reason, will require additional equipment in the amounts indicated.

An effort was also made to determine the extent of facilities and services to be provided at the launch base for payloads in the Shuttle system. These items would also be included and described in detail in the Shuttle Users Payload Support Handbook. The listings in Tables 2-3 and 2-4, representing an approach reflecting the general philosophy, should not be considered constraining or complete.

2.7 REFURBISHMENT

Refurbishment of the reusable payloads was investigated to determine the flow of operations and timelines involved. The following assumptions and ground rules were established.

(1) The payload incorporates the modular replacement concept, standardized equipment, and clean interface in the design.

(2) The payloads are similar or of a family type.

(3) Refurbishment restores the payload to original design life condition.

(4) Operations are planned for scheduled/unscheduled returns.

Table 2-3. Launch Base/Pad Area/Safing Area Provisions

Area	Facility/Activity	Service
Launch Pad Area	Mobile Service Tower/LUT	Vertical mating/de-mating of payload capability Payload quick-change capability Clean room - mated to orbiter payload bay
	Umbilical Tower (All payload functions go through orbiter)	Hard wire umbilicals Slave antennas and coax Payload cooling - air, cryo, coolant Propellant - load - offload capability Vent/drain - cryogens, effluents, propellants
Post-Landing Safing Area	P/L not Demated - Safing only - critical mission item removal O.K. Orbiter returns to orbiter operations facility for payload demating. P/L De-Mated - add to above, the equipment indicated in P/L Ground Support Equipment List	Ground power - AC-DC for payload Propulsion - defueling, flush, purge Cooling - air, cryogenic, coolant Pyrotechnic - system c/o, safing
General Launch Base	(See Facilities Descriptions, Kennedy Space Center/Air Force Eastern Test Range, TR-1080, Rev. 1, 15 May 71, by Space Shuttle Task Group, Center Planning and Future Programs, KSC NASA, Kennedy Space Center, Florida, 32899)	• motor pool • chemical lab. • calibration lab. • meteorology • photography • base medical facilities • base engineering shops • precision measurements lab

Table 2-4. Payload Support Facilities and Services

Facility	Service or Activity
*Administration	office and conference rooms
*Payload operations	pre-flight processing
*Payload storage	pre/post launch, controlled environment, monitored
*Flight crew	room(s) dedicated for payload crew
*Spares storage	bonded for AVE, AGE
*General equipment storage	consoles, mechanical shipping cases
Optics laboratory	repair, calibration, cleaning
Photo laboratory	repair, film storage, processing
Guidance and navigation laboratory	rate table
Battery laboratory	storage, activate, service, test
Clean laboratory	booths, benches, calibration sources
*Special laboratories	space provided for special use
Mechanical shop	general repair, maintenance, light fabrication
Electric/Electronics shop	general maintenance, repair, light fabrication
Gas storage	cylinders, Ne, He, Kr, N_2, O_2, etc.
Dangerous materials storage	radioactive calibration source, pyro, etc.
Propellant area	storage, disposal
(If payload refurbished on-site add:)	
Refurbish area	disassembly and assembly, test
Thermal-vacuum chamber	tie in to unified test equipment
Acoustic chamber	tie in to unified test equipment
Mass properties	weight, balance, spin
Vacuum chambers	black box & subsystem module capability

*May be assigned and dedicated to payload from activation through mission requirement.

(5) Complete spares, personnel, and payload-peculiar support equipment is available at the refurbishment location.

(6) Complete standardized test, checkout servicing equipment, and support is available at the refurbishment location and UTE or equivalent is used.

(7) The refurbishment location has acceptance test capability including thermal-vacuum and acoustic.

(8) Off-launch site location requires four additional days, two days each going and coming, for transportation preparation and air delivery.

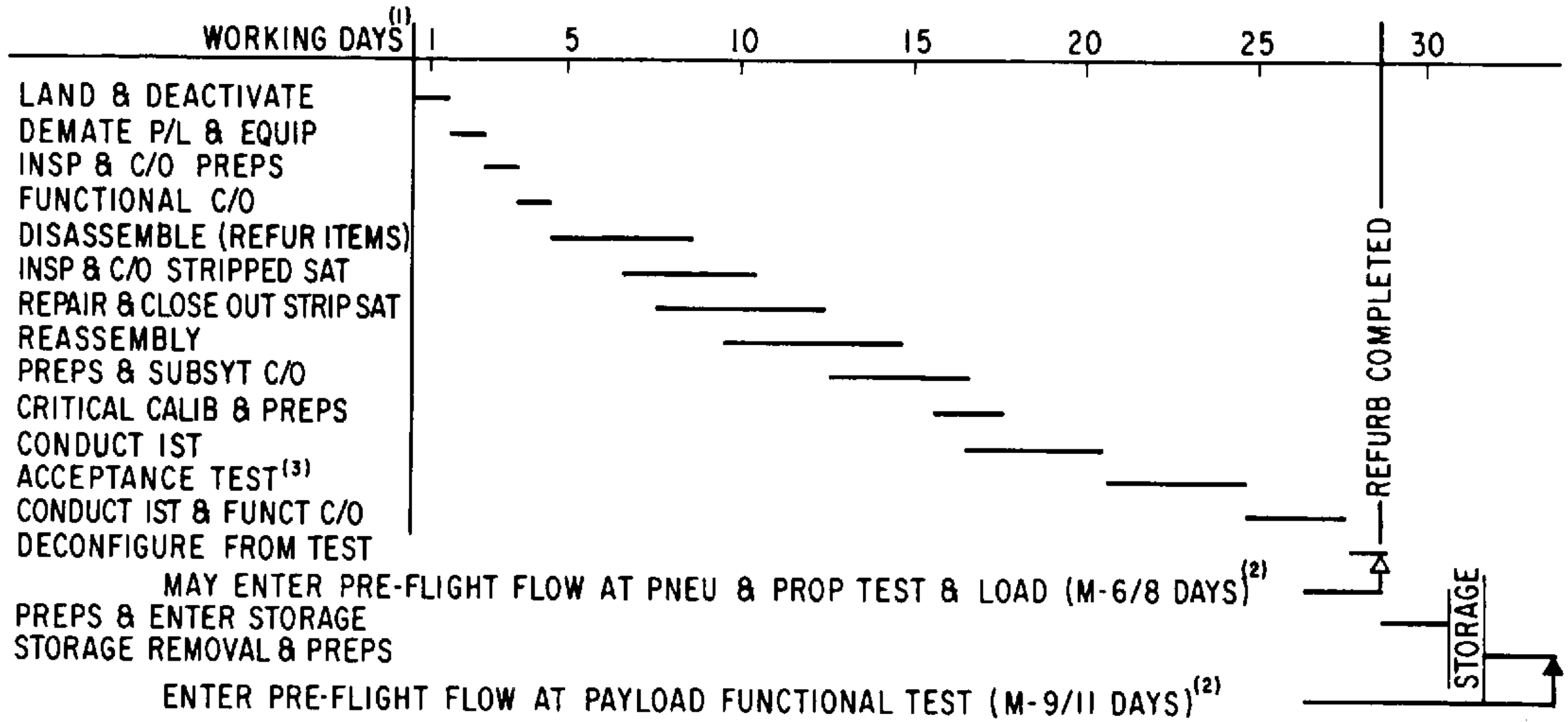
(9) There is no constraint on the location of the refurbishment site.

These ground rules and assumptions provided an approach to establishing a flow and timeline for the typical autonomous satellite that was developed. It is presented in Figure 2-16. It should be noted that by accomplishing the refurbishment operations at the launch site the satellite bypasses several of the operations normally scheduled for payloads (see Figure 2-2) arriving from off-site locations. This, together with the elimination of transportation time, has the effect of reducing reusable satellite refurbishment turnaround time.

2.8 OBSERVATIONS

Various current payload operation flows and contractors' studies on Shuttle/payload were reviewed and a detail baseline flow was developed for the Shuttle mode. Corresponding timelines were investigated and a synthesized timeline was also developed. It was observed from this task that the flow of operations and timelines for payloads presently appear to be relatively insensitive to the type of launch vehicle. This is indicated in comparing historical data with present contractor approaches to Shuttle operations. Current payload pre-launch operations are applied extensively to operations in the Shuttle era, and while the sequence and site may be varied, there are little or no functional differences.

APPROACH TO TYPICAL SATELLITE REFURBISHMENT



(1) NUMBER OF SHIFTS OPTIONAL AS REQUIRED

(2) M = ORBITER MATE DAY, ELAPSED TIME FOR ON-SITE MAINTENANCE FACILITY

(3) ADDITIONAL TIME WILL BE REQUIRED IF THERMAL-VACUUM TEST IS IMPLEMENTED

Figure 2-16. Typical Satellite Refurbishment Flow - Timeline

Using this baseline flow and synthesized timeline, the operations tradeoff analysis indicated that 10-day launch centers appear nominal for 20 launches per year per launch site. Servicing this launch rate, with 25 days to process a payload at the launch site, requires three bays. The three-bay concept does not include launch site refurbishment operations.

The L-1 mate day is preferred over the L-5 day because it will provide more time for payload processing and will be closer to the launch date, thereby minimizing dormancy failures and calibration drifts.

The flow of operations and corresponding timelines for payloads to be used in the Shuttle system can be reduced if the payload and associated support equipment and operations are specifically designed to take advantage of the capabilities offered by the Shuttle system. If this approach to standardization is adopted the operational times can be reduced substantially. Facilities, number of base personnel, payload contractor launch site support, refurbishment time, and dormancy failures will thereby be reduced. The list of potential standardized ground equipment, facilities, and provisions are provided. It should be recognized that improvement provided to the Shuttle system could also be provided to an expendable system. The significant advantages to be gained, however, are through integration with the single launch system utilized for all payloads.

2.9 REFERENCES

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2.3 OAO/LST Shuttle Economics Study, Grumman Aerospace Corporation (October 1970).

2.4 HEAO Phase-B Final Report, Volume II Detailed Briefing, and Volume IIIC Preliminary Test Requirements/Plan, TRW Systems Group (23 April 1971).

2.5 Shuttle System Impact Study, SD72-SH-0017, Final Report, Appendix 6, Ground Operations, North American Rockwell (March 1972).

2.6 Implementation of Research and Applications Payloads at the Launch Site, Detailed Technical Volume I and Appendix, Martin Marietta Co. (November 1971).

2.7 Shuttle Orbital Applications and Requirements (SOAR), Final Report Technical Volume, McDonnell Douglas Astronautics Corp. (October 1971).

3. Tug Operations

3. TUG OPERATIONS

3.1 INTRODUCTION

Analyses of Tug and tandem Tug performances have been performed to determine the timelines and velocities. Table 3-1 shows the Tug missions for which timelines and velocities have been determined. Timelines for synchronous equatorial missions with single and tandem Tugs were investigated; single Tug trajectories were examined for 12-hr orbits and 110-deg inclination.

The payload weights for these missions were based on weights from the 1971 NASA mission model. For instance, the 4,535-kg (10,000-lb) sync equatorial represents the Application Technology Satellite class payload, the 635-kg (1,400-lb) sync equatorial represents the Comsat class, and the 454-kg (1,000-lb) sync equatorial represents the synchronous meteorological type satellites.

The single Tug configuration assumed that the Tug and payload are assembled on the ground. The tandem Tug configuration assumed that the first Tug was launched first and the second Tug and payload were launched 24 hours later. The first Tug and second Tug/payload made rendezvous and docked to form the tandem Tug in a 185 x 185-km (100 x 100-nmi) x 28.5 deg orbit.

The Tug characteristics utilized in this study were based on the McDonnell Douglas OOS configuration (Ref. 3.1). This upper stage gross weight including a 4,535-kg (10,000-lb) payload was 35,827 kg (79,000 lb) with 470-sec specific impulse. At the time this study was conducted, the MSFC Tug configuration (Ref. 1.3 and 1.4) was not available. This difference in configuration should not influence the timelines, but it will have a small influence on the velocities determined for the various maneuvers. The timelines should be representative. It should be recognized that in those

Table 3-1. Tug Missions Investigated

Mission Designation	Orbit	Mode	Payload Weight kg	Payload Weight (lb)	Tug Configuration
A	Synchronous Equatorial	Single deployment	4,535	10,000	Single & Tandem
B	Synchronous Equatorial	Single retrieval	1,587	3,500	Single & Tandem
C	33,618 x 37,876 km (18,172 x 20,474 nmi) x 5 deg	Multiple deployment	2 + 635	2 x 1,400**	Single
D	555 x 555 km (300 x 300 nmi) x 110 deg	Multiple* service	3,111 up 4,925 down	8,800 up 12,800 down	Single
E	Synchronous Equatorial	Multiple service	3 + 454	3 x 1,000***	Tandem
F	Synchronous Equatorial	Single retrieval	4,319	9,524	Tandem

Notes: *single payload up 3,111 kg (8,800 lb) and two payloads down 3,111 + 1,814 kg (8,800 + 4,000 lb)

**payloads spaced 180°

***payloads spaced 120°

missions where phasing is used and multiple payloads are deployed, retrieved, or serviced at various longitudes the times are a function of the allocated velocities for these maneuvers.

3.2 DEPLOY IN SYNCHRONOUS EQUATORIAL ORBIT (A)

3.2.1 Single Tug

3.2.1.1 Deploy and Checkout

The Tug is separated from the Shuttle and deployed in the 185-km (100-nmi) circular parking orbit. On-orbit checkout of the Tug is initiated and completed during the phasing orbit wait time. Since the nominal Shuttle injection occurs at approximately maximum declination in the southern hemisphere, the initial synchronous transfer injection opportunity occurs at the first ascending node. This happens, however, only 20 minutes after the earliest possible deployment. For this mission the first synchronous transfer injection opportunity is scheduled for the following descending node. This occurs approximately 1 hr after injection of the Shuttle into the 185-km (100-nmi) orbit and imposes the requirement on the Tug that the minimum time for on-orbit checkout be 1 hr. (See Table 3-2).

3.2.1.2 Ascent Phasing

If the Tug is to place the payload at a prescribed longitude in synchronous orbit, several phasing operations are required. The first operation is a phasing wait in the 185-km (100-nmi) parking orbit to permit arrival of synchronous altitude close to the prescribed longitude. In order to maximize payload capability, the minimum-energy Hohmann transfer has been adopted in the transfer to synchronous orbit. This transfer results in consecutive ascending (or descending) arrival nodes at synchronous orbit being separated by approximately 22.5 deg of longitude, the nodal longitude shift

Table 3-2. Mission Timeline A — Single Tug

Objective: Deploy 10,000-lb payload in synchronous equatorial orbit.

Operation	No. of Burns (Main Engine)	Velocity (fps) (Main Engine)	Time Required for Operation (hr)
1. Deploy and checkout Tug (100 × 100-nmi 28.5 deg orbit).	0	0	(a)
2. Phasing wait in 100 nmi	0	0	12.9 (b)
3. Establish 100 × 19,323-nmi transfer orbit with 2.2-deg plane change and transfer.	1	8,148	5.2
4. Establish near synchronous phasing orbit (17,399 × 19,323 nmi) with 25.74-deg plane change. Phase for one revolution.	1	5,627	22.4
5. Establish synchronous equatorial orbit at required longitude with an 0.56-deg plane change.	1	245	~0.1
6. Deploy payload and maintain orbit to achieve favorable transfer mode alignment.	0	0	15 (b)
7. Establish 19,323 × 100-nmi transfer orbit with 26.3-deg plane change and transfer.	1	5,872	5.2
8. Establish intermediate 100 × 4,262-nmi phasing orbit with a 1.15-deg plane change. Phase for one orbital revolution.	1	3,751	2.9 (b)
9. Establish macro-rendezvous with Shuttle (100 × 100 nmi) with a 1.05-deg plane change.	1	4,397	~0.1
10. Micro-rendezvous with Shuttle (attitude control system for propulsion)	0	0 561 (d)	~2 (c)
Totals	6	28,601	66 (b)

Notes:

(a) Minimum available time increment between Shuttle deployment and first nodal opportunity is approximately 1 hr. Checkout may be done during Operation 2.
(b) Worst-case values.
(c) Assumed.
(d) 2% reserves.

per orbit in the 185-km (100-nmi) parking orbit. By utilizing a maxi· um of 8.75 revolutions in the parking orbit (8 revolutions after the first a ' ıd-ing node opportunity previously described), synchronous arrival posit. can be attained that will position the Tug within 11.3 deg of any desired longitude. The maximum parking orbit phasing wait is then approximately 12.9 hr. The several methods of achieving the final 11.3 deg of longitudinal positioning are discussed in the following section. Low earth phasing orbits would significantly shorten the maximum duration of the activities described in section 3.2.1.4; high-altitude phasing orbits were adopted, however, and the 12.9-hr maximum time applied.

3.2.1.3 Ascent Transfer Orbit

There are several schemes that would provide the phasing required for this profile. Two that are compatible with the minimum energy Hohmann transfer have been considered. These are low altitude, outer phasing orbits and high altitude, inner phasing orbits. To better explain these phasing orbit alternatives, the Hohmann transfer maneuver is first described.

At the desired node, an impulse velocity is applied to produce a small inclination change and to raise apogee to synchronous altitude. At apogee, the impulsive velocity required to rotate the orbit through the remainder of the plane change and to inject into the synchronous orbit is added. The plane change split between apogee and perigee is optimized to minimize the velocity requirement.

If this minimum velocity expenditure is to be retained, the maneuvers to establish the phasing orbits must be provided in either apogee or perigee injection maneuvers. Low altitude, outer phasing orbits can satisfy this requirement. Reducing the impulsive velocity added at perigee while retaining the same yaw angle results in an orbit with a perigee altitude of 185 km (100 nmi), an apogee altitude much lower than synchronous, and smaller inclination change. The size of the phasing orbit is governed by the magnitude of the longitudinal node shift desired. Since these outer phasing orbits only decrease the longitude of the node, the maximum shift must be the full 22.5-deg longitudinal gap between nodes. This maximum shift results in a 185 x 7,855-km (100 x 4,262-nmi) phasing orbit with a period of approximately 2.9 hr, twice the period of the 185-km (100-nmi) parking orbit. After one revolution in the phasing orbit, perigee is again reached and the Hohmann transfer perigee injection is completed. There would be some apsidal rotation during the transfer but this would be negligible. The apogee injection would then be made at the required longitude. The major advantage of this mode is that the maximum phasing time is less than 13 hr.

The phasing orbit mode selected for this profile is a high altitude, inner phasing orbit. These orbits have apogees at synchronous altitudes and are attained in a manner directly analogous to the low altitude phasing orbits just discussed, but they occur at apogee of the transfer. The details of this mode are described for the maximum node shift case in the Phasing Orbit description below. The major advantages of this scheme are ease of navigation and the reduction in midcourse maneuver requirements. The major drawback is that this mode requires approximately one day longer to reach synchronous orbit than the low altitude phasing orbit mode.

When the desired ascending or descending parking orbit node is encountered, the injection into the synchronous transfer orbit is achieved by a single burn of the Tug main engine, which produces an impulsive velocity of 2,485 m/sec (8,148 ft/sec). This velocity is applied at a yaw angle of approximately 9 deg to reduce the orbit inclination by 2.2 deg and produce an elliptic orbit with perigee at parking orbit altitude and apogee at synchronous altitude. The time to transfer from 185 to 35,748 km (100 to 19,323 nmi) is 5.2 hr.

3.2.1.4 Phasing Orbit

The longitude of the arrival node at synchronous orbit is within 22.5 deg of its desired location. An inner phasing orbit is chosen that will position the node at the desired location. The maximum node shift of 22.5 deg (this inner phasing orbit decreases the node longitude) is achieved by utilizing a 32, 188 x 34, 748-km (17, 399 x 19, 323-nmi) orbit inclined at 0.56 deg to the equator. To achieve this orbit, an impulsive velocity increment of 1, 216 m/sec (5, 627 ft/sec) must be applied at a yaw angle of 49.5 deg. The Tug coasts in this phasing orbit for 22.5 hr.

3.2.1.5 Injection Into Payload Orbit

After one revolution in the phasing orbit, the desired longitudinal position has been attained and the injection into synchronous equatorial orbit is performed. An impulsive velocity of 75 m/sec (245 ft/sec) added at a yaw angle of 23.8 deg is required for this maneuver.

3.2.1.6 Deploy Payload

The payload is separated from the Tug and 3 hr are allotted for payload checkout. The Tug remains in synchronous orbit until the time the return transfer orbit perigee (again assuming a Hohmann transfer) occurs at the ascending or descending node of the Shuttle parking orbit. The maximum wait including one-half the synchronous orbit period totals 15 hr.

3.2.1.7 Descent Transfer Orbit

When the correct nodal alignment described in section 3.2.1.6 has been achieved, the return transfer from synchronous orbit to the Shuttle parking orbit is initiated. This maneuver, which includes a 26.3-deg plane change, requires a velocity expenditure of 1, 791 m/sec (5, 872 ft/sec) applied at a yaw angle of 130.5 deg. The transfer itself consumes approximately 5.2 hr.

3.2.1.8 Return Phasing Orbit

When the Tug arrives at the parking orbit node, the Shuttle will probably be at some other position in the orbit and an in-plane true anomaly phasing maneuver will be required. Since the parking orbit is at such a low altitude, the elliptical phasing orbit must be an outer phasing orbit. The maximum phasing angle to be removed can approach 360 deg. To accomplish this maximum phasing condition in one revolution of the phasing orbit, the apogee altitude would have to be chosen to provide a period twice the 185-km (100-nmi) circular orbit period. That apogee altitude is 7, 885 km (4, 262 nmi). During the phasing wait in this orbit there is a relative node shift between the Tug and Shuttle because of differing orbit regression rates. Since the phasing relationships will be known before the synchronous deorbit maneuver, the departure longitude will be biased to account for this node shift. There will also be some apsidal rotation in the phasing orbit but this will be small (0.4 deg) in one revolution, and the altitude change at macro-rendezvous will be within rendezvous requirement accuracy. In order to minimize the velocity requirement, the plane change would be split between the injection into the phasing orbit and macro-rendezvous maneuvers. The velocity requirement to inject into the 185 x 7, 885-km (100 x 4, 262-nmi) orbit at perigee is 1, 139 m/sec (3, 751 ft/sec) applied at a yaw angle of 170.9 deg. The Tug coasts in this orbit for 2.9 hr. If the true anomaly differential is less than 360 deg, the period of the phasing orbit and the apogee altitude will then be reduced.

3.2.1.9 Macro-Rendezvous With Shuttle

When the Tug has completed one revolution in the elliptical phasing orbit, it will be approaching the Shuttle at a relative velocity of up to 1, 342 m/sec (4, 400 ft/sec). The Tug performs an impulsive maneuver to cancel this velocity, leaving the Tug coplanar and coaltitude with and in the near vicinity of the Shuttle. The velocity required to perform the macro-rendezvous maneuver from the 185 x 7, 885-km (100 x 4, 262-nmi) phasing orbit is 1, 341 m/sec (4, 397 ft/sec) applied at a yaw angle of 172 deg.

3.2.1.10 Micro-Rendezvous With Shuttle

Safety considerations dictate that the macro-rendezvous maneuver terminate with Shuttle and Tug at a distance greater than a minimum specified separation distance. A micro-rendezvous and docking maneuver is required.

The velocity requirement for Mission Timeline A is 8,552 m/sec (28,040 ft/sec). A 2% reserve has been added for non-nominal performance, resulting in a total mission velocity of 8,723 m/sec (28,601 ft/sec). A total of six main propulsion system burns are required, and the total mission duration (deployment from the Shuttle to recovery by the Shuttle) is a maximum of 68 hr.

3.2.2 Tandem Tug

3.2.2.1 Deploy Tandem Tug

The tandem Tug is assembled and deployed in the 185-km (100-nmi) circular parking orbit. On-orbit checkout of the tandem Tug is initiated and is completed during the phasing wait described in section 3.2.2.2.

3.2.2.2 Ascent Phasing

If the Tug is to place the payload at a prescribed longitude in synchronous orbit, several phasing operations are required. The first is a phasing wait in the 185-km (100-nmi) parking orbit. The purpose of this phasing wait is to permit the arrival at synchronous altitude as close as possible to the prescribed longitude. Transfer to synchronous altitude is initiated at either the ascending or descending equatorial crossing (node) of the 185-km (100-nmi) parking orbit, depending on the final synchronous longitude. Consecutive nodes of the parking orbit are separated by approximately 22 deg of longitude.

The deployment accomplished in section 3.2.2.1 will occur at some positive latitude prior to the first descending node of the Shuttle orbit. For worst case situations (complete autonomy), the Tug must determine its position and attitude as well as the on-orbit checkout before it can be ready to leave for synchronous altitude. Thus, the first opportunity assumed available will be the first ascending node of the Shuttle orbit.

In view of these facts, the possible arrival longitudes at a synchronous altitude are separated by 22 deg of longitude. By utilizing a maximum of eight revolutions in the parking orbit, synchronous arrival positions can be attained that will place the Tug within 11 deg of any desired final longitudinal position. Thus, the maximum parking orbit phasing wait is about 12.9 hr.

3.2.2.3 Tandem Transfer Orbit

The transfer to synchronous orbit altitude is accomplished by burning both stages. The first stage boosts the second stage and payload into an orbit with a 13,163-km (7,115-nmi) apogee. The staging is assumed to take place in the first revolution (prior to apogee) of the boost orbit. The second stage coasts to perigee and then continues to burn until the desired apogee conditions are attained (see section 3.2.2.7). The first stage remains in the boost orbit and coasts.

3.2.2.4 Descent Plane Change for Tug-1

The Tug-1 remains in the boost orbit and coasts to the intersection of the Shuttle and boost orbit planes where a small plane change is made to account for the relative nodal regression between the Shuttle and Tug-1. This amounts to 97 m/sec (317 ft/sec)(1.78-deg plane change) and includes the plane rotation of a 180-deg in-plane phasing orbit.

3.2.2.5 Descent Phasing Orbit for Tug-1

The Tug-1 now coasts to perigee and injects into a phasing orbit that will place Tug-1 in the near vicinity of the Shuttle. In order to have a 180-deg in-plane phasing angle capability, the phasing orbit would require a period of 1.5 times the Shuttle orbit period. This orbit would have an apogee altitude of 4,261 km (2,303 nmi) and require a perigee velocity increment of 869 m/sec (2,848 ft/sec).

3.2.2.6 Parking Orbit for Tug-1

After one revolution in the phasing orbit, Tug-1 breaks to the Shuttle orbit. The phasing orbit can be adjusted to permit any lead or lag angle desired with respect to the Shuttle. Tug-1 will now remain in this orbit until Tug-2 returns from the synchronous orbit altitude and effects a rendezvous with it (see section 3.2.2.14, below).

3.2.2.7 Ascent Transfer Orbit

In terms of time, this operation follows the separation of Tug-1 and Tug-2 and consists in the second stage coasting through apogee of the boost orbit to perigee, where it continues the ascent to synchronous orbit altitude. An alternative that could be used is to allow the second stage to coast to apogee, and then transfer to synchronous altitude (bielliptic transfer). For the case considered, however, an orbit with an apogee altitude of 35,748 km (19,323 nmi) and a perigee of 185 km (100 nmi) is established. A 2.2-deg plane change is made with this maneuver.

3.2.2.8 Phasing Orbit

As previously described, arrival at synchronous altitude can be attained to within 11 deg of any desired longitudinal position. This means that additional phasing may be required to reach the desired longitude. If 11 deg of phasing are required, the phasing can be completed in approxima' 'y one day by injecting into a near synchronous orbit. This orbit will have a period either slightly greater or slightly less than the synchronous orbit period.

The worst-case situation is shown in Table 3-4. It consists of overinjecting at the synchronous orbit altitude to establish a phasing orbit with a period of 24.8 hr. This would advance the longitude position of the satellite 11 deg per revolution. Retarding the longitudinal position 11 deg per revolution requires an underinjection resulting in an orbit with a synchronous altitude apogee and a perigee altitude of 34, 175 km (18, 457 nmi) with a period of 23.3 hr.

Since the plane change angle was split during the ascent, a plane change angle of 28.5 deg of relative inclination is removed in conjunction with establishing the near synchronous phasing orbit. The velocity required for this maneuver is 1, 814 m/sec (5, 976 ft/sec).

3.2.2.9 Deploy Payload

After one revolution in the near synchronous phasing orbit, the desired longitudinal position is attained. At this time a maneuver requiring 38 m/sec (114 ft/sec) is performed. This results in the final synchronous equatorial orbit. The payload is separated from Tug-2.

3.2.2.10 Phasing Orbit

Tug-2 remains in synchronous orbit until the return transfer perigee occurs at either an ascending or descending node of the Tug-1 parking orbit. This phasing wait will be a maximum of 12 hr. It is the reverse problem of the phasing wait described in section 3.2.2.2.

3.2.2.11 Descent Transfer Orbit

When the nodal alignment phasing described in section 3.2.2.10 is complete, the return transfer from synchronous altitude to the Tug-1 parking orbit altitude is initiated. This maneuver, including a 26.3-deg plane change, requires a velocity increment of 1, 788 m/sec (5, 862 ft/sec). The transfer time is 5.3 hr.

3.2.2.12 Phasing Orbit Near Tug-1

When Tug-2 arrives at the parking orbit node position, Tug -1 will probably be at some position in the parking orbit other than the node, and an in-plane phasing maneuver will be required. Since the parking orbit is at such a low altitude, the elliptical phasing orbit will have to be an outer phasing orbit (apogee greater than parking orbit altitude). Allowance has, therefore, been made for a phasing angle of up to 180 deg that might have to be removed. To accomplish this phasing in one revolution of the phasing orbit, the phasing orbit apogee altitude would have to be selected such that the phasing period is very nearly 1.5 times the parking orbit period. This corresponds to an apogee altitude of 4,261 km (2,303 nmi). Including the remaining 2.2-deg plane change, a velocity increment of 1,765 m/sec (5,786 ft/sec) opposing the velocity vector direction is applied immediately upon the arrival of Tug-2 at the parking orbit node. The one revolution in the phasing orbit takes 2.2 hr. There is no velocity penalty associated with this rendezvous technique (from synchronous orbit) since outer phasing orbits with respect to the target orbit are used. If the phasing angle to be removed is less than 180 deg, the time required for the phasing operation as well as the velocity required will decrease.

3.2.2.13 Tug-1 and Tug-2 Rendezvous

When Tug-2 has completed one revolution in the elliptical phasing orbit, it will be in rendezvous position with Tug-1. The Tug-2 phasing orbit is circularized at this time, thereby theoretically decreasing to zero the relative velocity between Tug stages. This maneuver requires as additional velocity increment of 878 m/sec (2,879 ft/sec) in opposition to the velocity vector. This completes the macro-rendezvous between the Tug stages.

There will be an error in the macro-rendezvous position and velocity. This is due to non-nominal performance in the propulsion and attitude control systems and uncertainties in both the Tug-1 and Tug-2 orbits. A micro-rendezvous and docking maneuver is therefore required. It is assumed in this analysis that the propulsion for this maneuver is provided by the Tug attitude control system, and the velocities are not included in the total mission velocity. An arbitrary estimate of two orbits was made in this analysis for the micro-rendezvous and docking operation.

3.2.2.14 Shuttle Rendezvous

Since the Tug-1/Tug-2 rendezvous is conducted in the near vicinity of the Shuttle, only a micro-rendezvous is required between the Shuttle and the Tug.

The velocity required for the tandem stage operation of Mission 1 is 9,766 m/sec (35,020 ft/sec). An additional 213 m/sec (698 ft/sec), representing a 2% flight performance reserve (FPR) for non-nominal performance in each stage, results in the total mission velocity of 10,894 m/sec (35,718 ft/sec). A maximum of six main propulsion system burns is required in the second stage and four in the first stage. The mission duration (Tug deployment to Tug recovery) is 3.02 days and is shown in Table 3-3.

3.3 RETRIEVE SMALL PAYLOAD FROM SYNCHRONOUS EQUATORIAL ORBIT (B)

3.3.1 Single Tug

This profile is identical to Mission Timeline A, Single Tug, except that Operations 6 and 7 consist of a micro-rendezvous and docking maneuver with the satellite, and phasing in synchronous orbit rather than payload deployment and phasing only. Details of this operation are presented in Table 3-4.

3.3.2 Tandem Tug

This mission consists in retrieving a payload from a synchronous equatorial orbit with a tandem Tug. Operationally this mission is identical to Mission

Table 3-3. Mission Timeline A - Tandem Tug

Objective: To deploy a 3,111-kg (8,800-lb) payload in synchronous equatorial orbit.

Operation	No. of Burns (Main Engine)		Velocity (fps)		Time Required for Operation (hr)
	Tug-1	Tug-2	Tug-1	Tug-2	
1. Deploy and checkout Tug (100 × 100 nmi × 28.5 deg).					(a)
2. Wait in 100-nmi orbit for proper departure point.					12.9(b)
3. Establish boost orbit (100 × 7,115 nmi × 28.5 deg).	1		5,719		~0.1
4. Tug-1 separates and coasts to intersection of Shuttle orbit plane and performs node shift to realign orbit planes.	1		317		4.1
5. Coast to perigee and establish phasing orbit with Shuttle (100 × 2,303 nmi × 28.5 deg).	1		2,848		
6. Break to Shuttle orbit and coast (100 × 100 nmi × 28.5 deg).	1		2,871		2.2
7. Second stage coast to perigee and establish transfer orbit to synchronous altitude with 2.2-deg plane change (100 × 19,323 nmi × 26.3 deg).		1		2,657	4.1

(a) Can be accomplished within the phasing wait of Operation 2, if tandem Tug is assembled.
(b) Worst-case value.
(c) 180-deg phasing angle allowed.
(d) Micro-rendezvous assumed accomplished with APS unit.

Table 3-3. Mission Timeline A - Tandem Tug (Continued)

Operation	No. of Burns (Main Engine)		Velocity (fps)		Time Required for Operation (hr)
	Tug-1	Tug-2	Tug-1	Tug-2	
8. Coast to apogee and establish high altitude phasing orbit for final payload placement with 26.3-deg plane change (19,323 × 20,379 nmi × 0 deg).		1		5,976	5.2
9. Establish final orbit and deploy payload (19,323 × 19,323 nmi × 0 deg).		1		114	24.8
10. Coast for Shuttle nodal alignment return opportunity (19,323 × 19,323 nmi × 0 deg).					12 (b)
11. Deorbit to Shuttle altitude with 26.3-deg plane change angle and coast to perigee (100 × 19,323 nmi × 26.3 deg).		1		5,862	5.2
12. Establish Shuttle phasing orbit with 2.2-deg plane change angle (100 × 2,303 nmi × 28.5 deg).		1		5,786	2.2
13. Break to Shuttle orbit and rendezvous with first stage.		1		2,870 (d)	3.0
14. Rendezvous with first stage in Shuttle orbit, then with Shuttle.				(d)	3.0
Subtotal	4	6	11,755	23,265	72.5
2% FPR			234	464	
Total	4	6	11,989	23,729	72.5

Table 3-4. Mission Timeline B — Single Tug

Objective: Non-time-critical retrieval of a 3,500-lb payload from synchronous equatorial orbit.

Operation	No. of Burns (Main Engine)	Velocity (fps) (Main Engine)	Time Required for Operation (hr)
1. Deploy and checkout Tug (100 × 100-nmi 28.5-deg orbit).	0	0	(a)
2. Phasing wait in 100-nmi orbit.	0	0	12.9
3. Establish 100 × 19,323-nmi transfer orbit with 2.2-deg plane change. Coast to apogee.	1	8,148	5.2
4. Establish near-synchronous phasing orbit (17,399 × 19,323 nmi) with 25.74-deg plane change and phase for one revolution.	1	5,627	22.4
5. Establish synchronous equatorial orbit at required longitude with 0.56-deg plane change.	1	245	0.1
6. Perform micro-rendezvous with satellite (attitude control system for propulsion).	0	0	12[c]
7. Phase for correct node alignment.	0	0	12[b]
8. Inject into 19,323 × 100-nmi transfer with 26.3-deg plane change and coast to perigee.	1	5,872	5.2
9. Establish intermediate 100 × 4,262-nmi orbit with a 1.15-deg plane change. Phase for one revolution.	1	3,751	2.9[b]
10. Perform macro-rendezvous with Shuttle (100 × 100 nmi) with a 1.05-deg plane change.	1	4,397	0.1
11. Micro-rendezvous with Shuttle (attitude control system for propulsion)	0	0 567[d]	2[c]
Totals	6	28,601	75 hr

(a) Minimum time available before 1st burn is approximately 1 hr. Performed during phasing wait. (Operation 2).
(b) Worst-case values.
(c) Assumed.
(d) 2% reserves.

Timeline A, Tandem Tug with the exception of an additional function. This function is performed in Operation 9 and consists of rendezvous, docking, and retrieval of the payload. (See Tables 3-3 and 3-5).

Since the configuration is lighter at liftoff, Tug-1 will have a higher boost orbit apogee altitude of 17,964 km (9,710 nmi). The subsequent phasing orbits and velocity increments are correspondingly changed. The total velocity required to perform this mission is 10,948 m/sec (35,896 ft/sec) including the 2% FPR. The mission duration is 3.03 days.

3.4 DEPLOY IN 24-HR ORBIT WITH SINGLE TUG (C)

The objective of this mission is to deploy two 635-kg (1,400-lb) satellites into a 33,617 x 37,875-km (18,172 x 20,474-nmi) 24-hr orbit inclined at 5 deg to the equator. These satellites are to be separated in the orbit by a 180-deg true anomaly differential. (See Table 3-6).

3.4.1 Deploy and Checkout

Same as Operation 1 of Mission Timeline A - Single Tug Stage.

3.4.2 Phasing Orbit

The capability to place the satellites in orbit with an arbitrary node location has been provided. Several phasing operations are required to implement this capability. The first operation is a phasing wait in the 185-km (100-nmi) parking orbit to permit arrival at the point of injection into the final orbit close to the desired nodal longitude. In order to minimize stage size a Hohmann transfer has been adopted for the transfer. This transfer results in consecutive ascending (or descending) arrival nodes at synchronous orbit being separated by approximately 22.5 deg of longitude, the nodal longitude shift per orbit in the 185-km (100-nmi) parking orbit. By utilizing a maximum of 8.75 revolutions in the parking orbit (8 revolutions after the first

Table 3-5. Mission Timeline B - Tandem Tug

Objective: Retrieve a 1,587-kg (3,500-lb) payload from synchronous equatorial orbit.

Operation	No. of Burns (Main Engine)		Velocity (fps)		Time Required for Operation (hr)
	Tug-1	Tug-2	Tug-1	Tug-2	
1. Deploy and checkout Tug (100 × 100 nmi × 28.5 deg).					(a)
2. Wait in 100-nmi orbit for proper departure point.					12.9(b)
3. Establish boost orbit (100 × 9,710 nmi × 28.5 deg).	1		6,524		0.1
4. First stage separates and coasts to intersection of Shuttle orbit plane and performs node shift maneuver to realign orbit planes.	1		455		5.3
5. First stage coast to perigee and establishes phasing orbit with Shuttle (100 × 2,303 nmi × 28.5 deg).	1		3,653 (c)		5.3
6. First stage breaks to Shuttle orbit and coasts (100 × 100 nmi × 28.5 deg).	1		2,871		2.2
7. Second stage coasts to perigee of boost orbit and establishes transfer orbit to synchronous altitude with 2.2-deg plane change (100 × 19,323 nmi × 26.3 deg).		1		1,993	5.3

(a) Can be accomplished within the phasing time of Operation 2, if tandem Tug is assembled.
(b) Worst-case value.
(c) 180-deg phasing angle allowed.
(d) Micro-rendezvous performed with APS unit.

Table 3-5. Mission Timeline B - Tandem Tug (Continued)

Operation	No. of Burns (Main Engine)		Velocity (fps)		Time Required for Operation (hr)
	Tug-1	Tug-2	Tug-1	Tug-2	
8. Coasts to apogee and injects into near-synchronous phasing orbit to rendezvous with payload with 26.3-deg plane change (19,323 × 18,876 nmi × 0 deg).		1		5,825	5.3
9. Establishes synchronous orbit and rendezvous with payload (19,323 × 19,323 nmi × 0 deg).		1		51 (d)	23.6
10. Coasts for Shuttle nodal alignment return opportunity.					12.0(b)
11. Deorbits to Shuttle altitude with 26.3-deg plane change and coasts to perigee (100 × 19,323 nmi × 26.3 deg).		1		5,875	5.3
12. Establishes phasing orbit with Tug-1 with 2.2-deg plane change (100 × 2,303 nmi × 28.5 deg).		1		6,057 (c)	2.2
13. Establishes Shuttle orbit and rendezvous with first stage (100 × 100 nmi × 28.5 deg).		1		2,870 (d)	3.0
14. Rendezvous with Shuttle.				(d)	3.0
Subtotal	4	6	13,503	22,671	72.7
2% FPR			270	452	
Total	4	6	13,773	23,123	72.7

Table 3-6. Mission Timeline C — Single Tug

Objective: Deploy two 1,400-lb satellites in a 18,172 × 20,474-nmi 5-degree inclined orbit.

Operation	No. of Burns (Main Engine)	Velocity (fps) (Main Engine)	Time Required for Operation (hr)
1. Deploy and checkout Tug (100 × 100-nmi 28.5-deg orbit).	0	0	(a)
2. Phasing wait in 100-nmi orbit.	0	0	10.0(b)
3. Establish intermediate phasing orbit (100 × 4,262 nmi for maximum phasing) with i = 27.3 deg and coast for one revolution.	1	4,398	2.9(b)
4. Establish 100 × 18,172-nmi transfer orbit i = 26.55 deg and coast to apogee.	1	3,616	4.9
5. Inject into 18,172 × 20,474-nmi orbit at i = 5 deg.	1	5,859	~0.1
6. Deploy satellite No. 1.	0	0	6
7. Establish a phasing orbit to deploy satellite No. 2 (18,172 × 27,714 nmi) and coast for 2 revolutions.	1	639(c)	60(c)
8. Establish final orbit for satellite No. 2.	1	639(c)	~0.1
9. Deploy satellite No. 2.	0	0	~0.1
10. Phase to Shuttle orbit plane intersection near apogee.	0	0	10
11. Inject into 100 × 20,296-nmi transfer orbit at i = 26.55 deg and coast to perigee.	1	5,350	6
12. Establish phasing orbit (100 × 4,262 nmi at i = 27.3 deg) and coast for one revolution.	1	3,825	2.9(b)
13. Perform macro-rendezvous with Shuttle (100 × 100 × 28.5 deg)	1	4,398	~0.1
14. Micro-rendezvous with Shuttle (ACS for propulsion).	0	0 574(d)	~2
Totals	8	29,298	105 hr or 4.3 days

(a) Minimum time available before first burn is approximately 1 hr. Performed during phasing wait.
(b) Worst-case value.
(c) Trade between time and velocity.
(d) 2% velocity reserve.

descending node opportunity) synchronous arrival positions can be attained that will position the Tug within 11.3 deg at any desired longitude. The maximum parking orbit phasing wait would be 12.9 hr. This accuracy is not sufficient for the profile under consideration. Both low altitude outer and high altitude inner phasing orbits are compatible with the Hohmann transfer. The low altitude orbit decreases the longitude of the arrival node and could require a longitudinal node shift of up to 22.5 deg. The period associated with this maximum shift is twice the parking orbit period and corresponds to an apogee altitude of 7,855 km (4,262 nmi) for a 185-km (100-nmi) perigee. With this technique the maximum waiting time in the parking orbit is 12.9 hr.

3.4.3 Intermediate Phasing Orbit

A low altitude outer phasing orbit has been selected for attaining the desired node locations of the payload orbit. As previously discussed, the maximum phasing orbit apogee altitude is 7,855 km (4,262 nmi). To minimize the total velocity requirement the plane change is split between maneuvers performed at perigee of the transfer orbit (1.95 deg) and the injection at apogee of this transfer orbit (21.55 deg). For the phasing orbit chosen for this representative profile, 185 x 7,855-km (100 x 4,262-nmi) orbit, a plane change of 1.2 deg is included in the perigee injection to establish the phasing orbit. An impulsive velocity of 1,341 m/sec (4,398 ft/sec) applied at a yaw angle of 9 deg is required to establish this orbit and rotate it through the desired plane-change angle. This maneuver is performed at a node. The Tug payload combination then coasts in this phasing orbit for one revolution.

3.4.4 Transfer Orbit

After one revolution in the phasing orbit the Tug injects at perigee into the transfer ellipse that intersects the desired payload orbit at apogee of the transfer ellipse. The perigee of the transfer orbit is the 185-km (100-nmi) parking orbit. Apogee can be at any point on the payload orbit. Preliminary analyses indicate the minimum total impulsive velocity for achieving the orbit is achieved if the payload orbit is oriented with its apogee at the apogee of the transfer orbit. (For this analysis it has been assumed that the payload orbit has its apsides at the orbit nodes.) If the apogee of the transfer orbit is located at any other point on the payload orbit, the total velocity requirement is increased. At perigee this increase amounts to approximately 110 m/sec (360 ft/sec). The apogee of the transfer orbit is then at an altitude of 33,618 km (18,172 nmi).

An impulsive velocity of 1,103 m/sec (3,616 ft/sec) is added at a yaw angle of 7.8 deg at perigee of the 368 x 7,855-km (199 x 4,262-nmi) phasing orbit to establish the transfer orbit. The Tug then coasts for 4.9 hr to reach apogee.

3.4.5 Inject Into Payload Orbit

At apogee of the transfer orbit, the Tug injects into perigee of the payload orbit. The impulsive velocity requirement for this injection is 1,787 m/sec (5,859 ft/sec) applied at a yaw angle of 41.7 deg.

3.4.6 Deploy First Payload

The first of the two satellites is deployed immediately following the injection maneuver.

3.4.7 Phasing Orbit For Second Payload

The Tug then injects into a phasing orbit to provide the required 180-deg true anomaly differential. This injection can again be performed at any point on the payload orbit. Two cases were considered for this analysis: apogee and perigee. The cost of establishing an inner and an outer phasing orbit at each was evaluated for a 60-hr phasing period. The results presented in Table 3-6 indicate that the outer phasing orbit with injection at the payload orbit perigee is the least expensive of the combinations considered. This mode was then adopted for this profile.

The impulsive velocity required to perform this phasing maneuver is a function of time allotted to perform the maneuver. Since there are no criteria for selecting this time, the 60-hr period was selected as being representative, and was used in this profile.

Immediately after separating the first payload the Tug adds an impulsive velocity increment of 195 m/sec (639 ft/sec) along the velocity vector establishing the 33,618 x 51,271-km (18,172 x 27,714-nmi) phasing orbit. The Tug then coasts for two revolutions on this 30-hr period orbit.

3.4.8 Inject Into Payload Orbit

At the completion of the second revolution in the payload orbit, the Tug injects into the payload orbit. At this time the first satellite has completed two and one-half revolutions in the payload orbit and is at apogee 180 deg ahead of the Tug. This injection is a retrograde maneuver requiring an impulsive velocity increment of 196 m/sec (639 ft/sec).

3.4.9 Deploy Second Payload

The second satellite is then deployed. The Tug, free of payload, coasts in the payload orbit until the correct geometry for transfer to the Shuttle low earth parking orbit occurs.

Again, there are several transfers that could be used to reach the desired low earth orbit. The "direct transfer" selected for this profile is not necessarily optimum, but it is simple and does not entail large phasing tir This "direct transfer" is defined as the intersection of the Shuttle a: .oad orbit planes. This line of nodes, referred to here as the plane line, intersects both Shuttle and payload orbits at two locations. The intersection of the plane line with the payload orbit near apogee is chosen as the departure point for the transfer orbit since it is the first to occur after the second payload deployment. The intersection of the Shuttle orbit that is most nearly antipodal to the departure point becomes perigee of the transfer orbit.

The phasing time in the payload orbit required to achieve the departure condition for the mission times assumed is approximately 10 hr.

3.4.10 Phasing Orbit

The Tug injects into the transfer orbit at an altitude of 37, 555 km (20, 300 nmi) where the payload orbit flight path angle is approximately 1.52 deg. A velocity increment of 1, 651 m/sec (5, 380 ft/sec) is applied at a yaw angle of approximately 161 deg to establish the transfer ellipse. Since the flight path angle is positive, apogee of the transfer orbit has not been reached. The coast time in this orbit is approximately 6 hr.

3.4.11 Ascent Transfer Orbit

The discussion presented for Operation 8 of Mission Timeline A - Single Tug is directly applicable to this operation. Only the values of the injection parameters vary.

The velocity requirement for injection into the 185 x 7, 855-km (100 x 4, 262-nmi) orbit is 1, 167 m/sec (3, 825 ft/sec) applied at a yaw angle of 170 deg.

3.4.12 Phasing Orbit

Same as Operation 9 of Mission Timeline A - Single Tug.

3.4.13 Macro-Rendezvous

Same as Operation 10 of Mission Timeline A - Single Tug.

The velocity requirement for Mission Timeline C is summarized in Table 3-6. A 2-percent reserve has been added, yielding a total mission velocity requirement of 8,936 m/sec (29,298 ft/sec). Eight main engine burns are required during the profile, which takes 4.3 days to accomplish.

3.5 SERVICE TO LOW SUN SYNCHRONOUS ORBIT (D)

3.5.1 Deploy and Checkout

A single Tug with a 3,991-kg (8,800-lb) package for servicing is separated from the Shuttle and deployed in a 185-km (100-nmi) circular orbit at 110-deg inclination. The Shuttle was launched in plane to minimize plane change requirements.

3.5.2 Phasing Orbit

The Tug is required to coast in the 185-km (100-nmi) orbit to attain the proper phase angle relationships with the satellite to be retrieved.

3.5.3 Transfer Orbit

The first maneuver is to establish a Hohmann transfer orbit to the vicinity of the target satellite. A velocity expenditure of 105 m/sec (315 ft/sec) and a transfer time of 0.8 hr is required.

3.5.4 Inject Into Payload Orbit

At apogee of the transfer orbit a 95-m/sec (311-ft/sec) impulse is added to circularize the orbit.

3.5.5 Macro-Rendezvous

The macro-rendezvous maneuver uses 28 m/sec (93 ft/sec).

3.5.6 Station-keeping

The Tug prepares itself for final rendezvous and docking by acquiring the proper approach heading and orientation.

3.5.7 Micro-Rendezvous and Dock

The final rendezvous and docking is accomplished arbitrarily, using one orbit for this purpose.

3.5.8 Phasing Orbit

In order to compensate for the difference in angular rate between the Tug and Shuttle orbits, a phasing orbit must be used. An impulse of 387 m/sec (1, 270 ft/sec) is added by the Tug to establish this orbit.

3.5.9 Transfer Orbit

A velocity of 98 m/sec (320 ft/sec) is used to lower the perigee radius to that of the Shuttle.

3.5.10 Macro-Rendezvous

The macro-rendezvous is completed by breaking the Tug and payload into the Shuttle orbit at 185 km (100 nmi).

3.5.11 Micro-Rendezvous

The micro-rendezvous is accomplished by using the ACS system. The detail timeline is shown in Table 3-7. The total time required is approximately 0.7 days including 0.75 hr of station-keeping time.

3.6 SERVICE TO SYNCHRONOUS EQUATORIAL ORBIT (E)

Table 3-7. Mission Timeline D — Single Tug

Objective: To carry a 3,111-kg (8,800-lb) package for servicing, and to retrieve a 1,814-kg (4,000-lb) satellite from a 555-km (300-nmi) circular orbit with an inclination of 110 deg.

Operation	No. of Burns (Main Engine)	Velocity (fps) (Main Engine)	Time Required for Operation (hr)
1. Deploy and checkout Tug. Performed during phasing wait (100/100/110 deg).	0	0	(a)
2. Phasing wait in 100-nmi orbit.	0	0	7.4
3. Establish transfer orbit (100 × 280 nmi).	1	315	0.8
4. Inject into circular orbit 280 × 110 deg.	1	311	0.1
5. Macro-rendezvous (300 × 300).	1	93	0.9
6. Stationkeeping.	0	0	0.45
7. Rendezvous and dock.	0[b]	0	0.5
8. Establish transfer orbit (300 × 1,176 nmi).	1	1,270	4.8
9. Perigee deboost 300 to 100 nmi.	1	320	0.9
10. Macro-rendezvous with Shuttle (100 × 100 × 110 deg).	1	1,620	0.5
11. Micro-rendezvous with Shuttle.	0[b]	79[c]	0.4
Totals		4,008	16.75

Notes:

(a) Minimum time before first burn ~1 hr.
(b) Accomplished with APS.
(c) 2% flight performance reserve.

3.6.1 Deploy and Checkout Tug-1

Tug-1 is separated from the Shuttle and deployed in the 185-km (100-nmi) circular parking orbit. On-orbit checkout of Tug-1 is initiated and completed during the orbit wait time of Operation 2.

3.6.2 Parking Orbit

Tug-1 coasts in orbit while the second Tug stage (Tug-2) with the payloads is readied and launched into a coelliptic orbit. The one-day time delay is an arbitrarily assumed Shuttle launch sequence.

3.6.3 Deploy and Checkout Tug-2 and Payload

Tug-2 with the three 454-kg (1,000-lb) payloads is separated from the Shuttle and deployed in the 185-km (100-nmi) circular parking orbit. On-orbit checkout of Tug-2 is initiated and completed during the phasing orbit wait time.

3.6.4 Macro-Rendezvous

Coincident with Operation 3, Tug-1 initiates phasing with Tug-2.

3.6.5 Dock

Tug-1 finalizes a rendezvous and docks with Tug-2.

3.6.6 Ascent Phasing Orbit

For the Tug to place the payload at a prescribed longitude in synchronous orbit, several phasing operations are required. The first is a phasing wait in the 185-km (100-nmi) parking orbit. The purpose of this phasing wait is to permit arrival at synchronous altitude as close as possible to the prescribed longitude. Transfer to synchronous altitude is initiated at either the ascending or descending equatorial crossing (node) of the 185-km (100-nmi) parking orbit, depending on the final synchronous longitude. The maximum parking orbit phasing wait is about 11.8 hr.

3.6.7 Transfer Orbit, Tandem Tug

The transfer to synchronous orbit altitude is accomplished by burning both stages. The first stage boosts the second stage and payload into an orbit with a 20,126-km (10,879-nmi) apogee. A 2.2-deg plane change is made with this maneuver. The staging is assumed to take place in the first revolution (prior to apogee) of the boost orbit. The second stage coasts to perigee and then continues to burn until the desired apogee conditions are attained (Operation 8). The first stage remains in the boost orbit and coasts.

3.6.8 Ascent Transfer Orbit, Tug-2

This operation in terms of time follows the separation of Tug-1 and Tug-2 and consists in the second stage coasting through apogee of the boost orbit to perigee where it continues the ascent to synchronous orbit altitude. An alternative that could be used is to allow the second stage to coast to apogee, and then transfer to synchronous altitude (bielliptic transfer). For the case considered, however, an orbit with an apogee altitude of 35,748 km (19,323 nmi) and a perigee of 185 km (100 nmi) is established.

3.6.9 Tug-1 Transfer Orbit

Tug-1 now coasts to perigee and injects into a phasing orbit that will place it in the near vicinity of the Shuttle.

3.6.10 Inject Into Synchronous Equatorial Orbit

Tug-2 injects into synchronous orbit when it arrives at apogee. Since the plane-change angle was split during the ascent, a plane-change angle of relative inclination is removed in conjunction with establishing the synchronous orbit. The velocity required for this maneuver is 1, 796 m/sec (5, 887 ft/sec).

3.6.11 Tug-1 Phasing Orbit

After one revolution in the phasing orbit Tug-1 breaks to the Shuttle orbit. The phasing orbit can be adjusted to permit any lead or lag angle desired with respect to the Shuttle.

3.6.12 Payload Deployments

These maneuvers consist in establishing the orbits of each individual payload to be deployed. Intermittent maneuvers are used to rendezvous and dock with the replaced satellites to be retrieved. The details are presented in Table 3-8.

3.6.13 Phasing Orbit

This operation is similar to the phasing required in Operation 6. Since a particular node is required, the opportunity to return occurs twice a day. Thus, a maximum of 12 hr may elapse before Tug-2 may return.

3.6.14 Descent Transfer Orbit

Tug-2 now transfers to 185 x 35, 748-km (100 x 19, 323-nmi) orbit. A 26.3-deg plane change is made with this maneuver.

3.6.15 Phasing Orbit

Tug-2 now makes a maneuver similar to that of Tug-1 in Operation 9 in order to phase with its Shuttle.

Table 3-8. Mission Timeline E - Tandem Tug

Ob Deploy and retrieve three 4,535-kg (1,000-lb) payloads from synchronous equatorial orbit.

Operation	No. of Burns (Main Engine)		Velocity (fps)		Time Required for Operation (hr)
	Tug-1	Tug-2	Tug-1	Tug-2	
1. Deploy and checkout Tug-1 (100 × 100 nmi × 28.5 deg).					
2. Wait in 100-nmi orbit for launch opportunity of Tug-2.					24
3. Deploy and checkout Tug-2 (100 × 100 nmi × 28.5 deg).					
4. Tug-1 injects to establish phasing orbit with Tug-2 and coasts (100 × 411 nmi × 28.5 deg).	1		533		23.5
5. Tug-1 breaks to Tug-2 orbit and to rendezvous with Tug-2 (100 × 100 nmi × 28.5 deg).	1		583		10
6. Wait in 100-nmi orbit for proper departure point.					11.8
7. Establish boost orbit with 2.2-deg plane change (100 × 10,879 nmi × 26.3 deg) and Tug-1 separates.	1		6,901		~0.1
8. Tug-2 coasts to perigee and establishes transfer orbit (100 × 19,323 nmi × 26.3 deg).		1		1,260	5.88

Table 3-8. Mission Timeline E - Tandem Tug (Continued)

Operation	No. of Burns (Main Engine)		Velocity (fps)		Time Required for Operation (hr)
	Tug-1	Tug-2	Tug-1	Tug-2	
9. Tug-1 coasts to perigee and establishes phasing orbit with Shuttle (100 × 411 nmi × 28.5 deg).	1		6,368		5.88
10. Tug-2 coasts to apogee and injects into synchronous equatorial orbit (19,323 × 19,323 nmi × 0 deg).		1		5,887	5.25
11. Tug-1 breaks to Shuttle orbit and performs rendezvous.	1		583		33.5
12. Tug-2 establishes phasing orbit for placement of first payload (18,390 × 19,323 nmi × 0 deg).		2		212	23.2
13. Deploy payload and rendezvous with satellite to be retrieved.		2		50	10
14. Establish phasing orbit to deploy payload No. 2 (19,323 × 28,948 nmi × 0 deg).		1		845	31.9
15. Establish payload No. 2 orbit (19,323 × 19,323 nmi × 0 deg).		1		845	~0.1
16. Deploy payload No. 2 and rendezvous with 2nd satellite to be retrieved.		2		50	10
17. Establish phasing orbit to deploy payload No. 3 (19,323 × 28,948 nmi × 0 deg).		1		845	31.9
18. Establish payload No. 3 orbit (19,323 × 19,323 nmi × 0 deg).		1		845	~0.1

Table 3-8. Mission Timeline E - Tandem Tug (Continued)

Operation	No. of Burns (Main Engine)		Velocity (fps)		Time Required for Operation (hr)
	Tug-1	Tug-2	Tug-1	Tug-2	
19. Deploy payload No. 3 and rendezvous with 3rd satellite to be retrieved.		2		50	10
20. Tug-2 waits for Shuttle nodal alignment return opportunity.					12.0
21. Deorbit to Shuttle altitude with 26.3-deg plane change (100 × 19,323 nmi × 26.3 deg).		1		5,887	5.25
22. Establish phasing orbit to Shuttle with 2.2-deg plane change (100 × 411 nmi × 28.5 deg).		1		6,369	23.5
23. Tug-2 breaks to Shuttle orbit and performs rendezvous.		1		583	10
Totals	5	17	4,565 m/sec (14,968 ft/sec)	7,237 m/sec (23,728 ft/sec)	#1 108.8 #2 224.6 overall 248.5

3.6.16 Rendezvous and Dock

Tug-2 finalizes its rendezvous with the Shuttle. The detail timeline is shown in Table 3-8. The total time including the deployment of three payloads is 10.3 days.

3.7 RETRIEVE LARGE PAYLOAD FROM SYNCHRONOUS ORBIT (F)

3.7.1 Deploy and Checkout Tug-1

Tug-1 is separated from the Shuttle and deployed in the 185-km (100-nmi) circular parking orbit. On-orbit checkout of Tug-1 is initiated and completed during the orbit wait time. Tug-1 coasts in orbit while Tug-2 is readied and launched into a coelliptic orbit. The one-day time delay is an arbitrarily assumed Shuttle launch sequence. (See Table 3-9.)

3.7.2 Deploy and Checkout Tug-2

Tug-2 is separated from its Shuttle and deployed in the 185-km (100-nmi) circular parking orbit. On-orbit checkout of Tug-2 is initiated and completed during the phasing orbit wait time.

3.7.3 Macro-Rendezvous - Tug-1

Coincident with Operation 2, Tug-1 initiates phasing with Tug-2.

3.7.4 Rendezvous and Dock

Tug-1 achieves a rendezvous and docks with Tug-2. For the Tug to place the payload at a prescribed longitude in synchronous orbit, several phasing operations are required. The first is a phasing wait in the 185-km (100-nmi) parking orbit. The purpose of this phasing wait is to permit arrival at synchronous altitude as close as possible to the prescribed longitude. Transfer to synchronous altitude is initiated at either the ascending or

Table 3-9. Mission Timeline F - Tandem Tug

Mission Objective: Retrieve a 4,535-kg (10,000-lb) payload from synchronous equatorial orbit.

Operation	No. of Burns (Main Engine)		Velocity (fps)		Time Required for Operation (hr)
	Tug-1	Tug-2	Tug-1	Tug-2	
1. Deploy Tug-1 into 100 × 100-nmi × 28.5-deg orbit.					1.6
Tug-1 coasts into 100 × 100-nmi × 28.5-deg orbit for launch opportunity of Tug-2.					24.0(a)
2. Launch Tug-2 into 100 × 100-nmi × 28.5-deg orbit, establish ephemeris.					1.6
3. Tug-1 injects into phasing orbit and coasts.	1		550		24.0(b)
4. Tug-1 recircularizes into 100 × 100-nmi × 28.5-deg orbit and effects micro-rendezvous with Tug-2.	1		700		3.0
Coast to departure point.					13.2(c)
5. Establish boost orbit (100 × 12,700 nmi × 28.5 deg) and separate Tug-1.	1		6,930		0.2

(a) Assumed
(b) Time - ΔV trade
(c) Worst-case values

Table 3-9. Mission Timeline F - Tandem Tug (Continued)

Operation	No. of Burns (Main Engine)		Velocity (fps)		Time Required for Operation (hr)
	Tug-1	Tug-2	Tug-1	Tug-2	
6. Tug-2 coasts to perigee and establishes transfer orbit with 2.2-deg plane change.		1		1,222	6.8
Tug-1 coasts to perigee and establishes phasing orbit (100 × TBD nmi × 28.5 deg).	1		6,380		6.8
7. Tug-2 coasts to apogee and injects into 19,323 × 19,323 ± 50-nmi × 0-deg orbit.		1		7,000	5.3
Tug-1 coasts to perigee, circularizes, and performs micro-rendezvous with Shuttle.	1		700		27.0(b)
8. Tug-2 coasts one revolution and performs micro-rendezvous with payload and docks (walking orbit).		1		155	27.0(b)
9. Tug-2 coasts for correct node alignment.					12.0(c)
10. Inject into 19,323 × 100-nmi × 26.3-deg orbit and coast to perigee.		1		5,872	5.2
11. Establish intermediate (4,262 × 100 nmi) with 1.15-deg plane change. Phase for one revolution.		1		3,751	2.9

Table 3-9. Mission Timeline F - Tandem Tug (Continued)

Operation	No. of Burns (Main Engine)		Velocity (fps)		Time Required for Operation (hr)
	Tug-1	Tug-2	Tug-1	Tug-2	
12. Perform micro-rendezvous with a 1.05-deg plane change.		1		4,397	0.1
Micro-rendezvous with Shuttle (ACS for propulsion).					2.0
Totals	5	6	15,262	16,097	#1 99.8 #2 103.3

descending equatorial crossing (node) of the 185-km (100-nmi) parking orbit, depending on the final synchronous longitude. The maximum parking orbit phasing wait is about 11.8 hr.

3.7.5 Transfer Orbit - Tandem Tug

The transfer to synchronous orbit altitude is accomplished by burning both stages. The first stage boosts the second stage and payload into an orbit with a 23,495-km (12,700-nmi) apogee. The staging is assumed to take place in the first revolution (prior to apogee) of the boost orbit. The second stage coasts to perigee and then continues to burn until the desired apogee conditions are attained (Operation 6). The first stage remains in the boost orbit and coasts.

3.7.6 Tug-1 and Tug-2 Transfer Orbit

This operation in terms of time follows the separation of Tug-1 and Tug-2 and consists in the second stage coasting through apogee of the boost orbit to perigee, where it continues the ascent to synchronous orbit altitude. An alternative that could be used is to allow the second stage to coast to apogee, and then transfer to synchronous altitude (bielliptic transfer). For the case considered, however, an orbit with an apogee altitude of 35,748 km (19,323 nmi) and a perigee of 185 km (100 nmi) is established. A 2.2-deg plane change is made with this maneuver.

The first stage now coasts to perigee and injects into a phasing orbit that will place Tug-1 in the near vicinity of the Shuttle.

3.7.7 Phasing Orbit

Tug-2 injects into synchronous orbit when it arrives at apogee. Since the plane-change angle was split during the ascent, a plane-change angle of 26.3 deg of relative inclination is removed in conjunction with establishing the synchronous orbit. The velocity required for this maneuver is 2,135 m/sec (7,000 ft/sec).

After one revolution in the phasing orbit, Tug-1 breaks to the Shuttle orbit. The phasing orbit can be adjusted to permit any lead or lag angle desired with respect to the Shuttle.

3.7.8 Rendezvous and Dock with Payload

This maneuver consists of minor phasing adjustments to complete the rendezvous with the target satellite.

3.7.9 Descent Phasing Orbit

This operation is similar to the phasing required in Operation 4. Since a particular node is required, the opportunity to return occurs twice a day. Thus, a maximum of 12 hr may elapse before Tug-2 may return.

3.7.10 Transfer Orbit

Tug-2 now transfers to 185 x 35, 748-km (100 x 19, 323-nmi) orbit. A 26.3-deg plane change is made with this maneuver.

3.7.11 Phasing Orbit

Tug-2 now makes a maneuver similar to that of Tug-1 in Operation 6 to phase with its Shuttle.

3.7.12 Rendezvous and Dock

Tug-2 finalizes its rendezvous with the Shuttle. The total timelines for Tug-1 and Tug-2 are 4.2 and 4.3 days for retrieval of large payloads with tandem Tug.

3.8 SUMMARY

Tl nber of Tug burns and velocities for single and tandem Tugs are su ized in Table 3-10 for the six missions. The data indicate that

Table 3-10. Summary of Mission Timelines

	MISSION DESIGNATION							
	A		B		C	D	E	F
MISSION ORBIT	Synchronous Equatorial		Synchronous Equatorial		18,172 x 20,474 n mi x 5°	300 x 300 n mi x 110°	Synchronous Equatorial	Synchronous Equatorial
PAYLOAD WEIGHT	Deploy 10,000 lb		Retrieve 3500 lb		Deploy Two 1400 lb Satellites 180° Apart	Deploy 8800 lb Retrieval 12,800 lb	Deploy and Retrieve Three 1000 lb Satellites	Retrieve 9524 lb
TUG CONFIGURATION	Single	Tandem	Single	Tandem	Single	Single	Tandem	Tandem
No. of Burns								
TUG-1	6	4	6	4	8	6	5	5
TUG-2	-	6	-	6	-	-	17	6
Time on Orbit (Hr)								
TUG-1	66	19.3	75	25.8	105	16.75	108.8	99.8
TUG-2	—	72.5	—	72.7	—	—	224.6	103.3
Total Velocity Rqmts (fps)								
TUG-1	28,601	11,989	28,601	13,773	29,298	4,008	14,968	15,262
TUG-2	—	23,729	—	23,123	—	—	23,728	16,097

the second Tug in the tandem arrangement results in the larger number of burns and time in orbit. The large number of burns results from the maneuvers for rendezvous with multiple payloads. The single Tug arrangement has a lesser number of burns and shorter on-orbit times. It should be recognized that the total velocity is based on the 1971-OOS version (Ref. 3.1) and should be viewed as comparative indicators between missions. The number of burns required will be the same as for the corresponding mission and trajectories with the MSFC Tug.

The Tug orbit timelines for the various missions were combined with the expected payload timelines and are shown in Tables 3-11 and 3-12 for single Tug and tandem Tug. In both cases only single payloads were considered. The range of values is given since the values are dependent on the payload checkout requirements and orbit phasing times. The lower value represents minimum values and the higher value represents maximum. The times are payload times and do not include descent times for deployment missions and ascent times for retrieval missions.

The maximum payload on-orbit times are summarized in Table 3-13 for deployment and retrieval. Those missions, without phasing, represent payloads not requiring longitudinal placement or rendezvous. Maximum retrieval times for single and tandem Tugs are the same because the return flight plans are the same. These maximum times can be of significance for those payloads that are deactivated for these time durations in the space environment. The payload during the transportation phase may be basically deactivated and only those items requiring appropriate environment and safety be activated. Time required for payload checkout on-orbit should be added to the timelines.

C-2

Table 3-11. Tug/Payload Timeline - Single Tug and Single Payload to Synchronous Equatorial Orbit

	TIME (Hrs)
● DEPLOY PAYLOAD (ASCENT)	
/ Deploy from Shuttle and Checkout Tug/Payload	1 to 3*
/ Phasing in 100 N Mi Orbit for Node Alignment	1 to 13
/ Hohmann Transfer to Synchronous Equatorial Orbit	5
/ Phasing to Mission Longitude (11.3^0 Phasing with Nominal ΔV)	0 to 24
/ Checkout, Activate and Deploy Payload	11 to 14*
TOTAL:	18 to 59
● RETRIEVE PAYLOAD (DESCENT)	
/ Dock, Checkout, Retract/Separate Appendages and Deactivate Payload	1 to 3*
/ Phasing for Node Alignment	0 to 12
/ Hohmann Transfer to Shuttle Orbit	5
/ Rendezvous with Shuttle	3 to 5
/ Checkout, Dock, Deactivate and Stow	1 to 4*
TOTAL:	10 to 29

* Estimates

Table 3-12. Tug/Payload Timeline - Tandem Tug and Single Payload to Synchronous Equatorial Orbit

	TIME (Hrs)
● DEPLOY PAYLOAD (ASCENT)	
/ Deploy from Shuttle, Rendezvous with Tug, and Check Out	2 to 28
/ Phasing in 100 N Mi Orbit for Node Alignment	1 to 13
/ First Tug Boost to Intermediate Transfer	4
/ Second Tug Boost to Synchronous Orbit	5
/ Phasing to Mission Longitude	0 to 24
/ Checkout, Activate, Deploy Payload	11 to 14*
TOTAL:	23 to 88
● RETRIEVE PAYLOAD (DESCENT)	
/ Dock, Checkout, Retract/Separate Appendages, and Deactivate Payload	1 to 3*
/ Phasing for Node Alignment	0 to 12
/ Hohmann Transfer to Shuttle Orbit	5
/ Rendezvous and Dock with Shuttle	3 to 5
/ Checkout, Dock, Deactivate and Stow	1 to 4*
TOTAL:	10 to 29

* Estimates

Table 3-13. Payload Design Impact from Tug Timelines - Single Payload to Synchronous Equatorial Orbit

	SINGLE TUG (HOURS)	TANDEM TUG (HOURS)
MAXIMUM DEPLOYMENT TIME		
WITH PHASING	59	88
WITHOUT PHASING	22	51
MAXIMUM RETRIEVAL TIME	29	29

- ELECTRICAL POWER AND ROLLING OF PAYLOAD (THERMAL CONTROL) MAY BE REQUIRED DURING ASCENT AND DESCENT TO MAINTAIN CRITICAL COMPONENTS WITHIN TEMPERATURE LIMITS

3.9 REFERENCE

3.1 Orbit-to-Orbit (Chemical) Feasibility Study, SAMSO-TR-71-221, McDonnell Douglas (October 1971).

4. Shuttle Sortie

4. SHUTTLE SORTIE

The design tradeoff study for sortie missions was conducted to provide data necessary to perform the cost trades for sortie, mission requirements, and manned accessibility to experiments. These were performed by configuring the mission equipment packages, developing equivalent solar observation programs, and designing various conceptual spacecraft.

To generate data for cost analysis on scientific data output, it was necessary to develop various traffic models using sortie, automated satellites, and a free flyer capable of providing equivalent solar programs. These equivalent programs were developed basically to compare solar programs over the 1979-1990 time period. The free flyer is not a sortie, but is included for comparison.

The mission requirement is assessed by conceptual designs of various sensor sizes and instrument pointing capabilities. The manned accessibility to instruments is a desirable feature as employed in the NASA CV-990 airborne laboratory activities. The scientists accompanying their experiment could perform on-flight repairs, adjustments, calibration, and experimentation aboard the aircraft. Design data to evaluate the costs and a gross operational evaluation for their various approaches are provided. The design data are in the form of conceptual layouts, subsystem descriptions, and weight estimates.

The data describing the orbiter and its characteristics were supplied in Ref. 4.1 through 4.4.

A sortie mission is defined for purposes of the conceptual design study as a short duration mission wherein the payload is retained with the Shuttle and is returned to earth by the orbiter at the conclusion of the mission.

4.1 SORTIE/SOLAR OBSERVATORY

For the sortie type mission, the large solar observatory was selected as the Shuttle-attached payload. This payload is a representative sortie mission since it is a candidate for an early astronomy experiment and is currently being studied for NASA/MSFC by The Aerospace Corporation's Laboratory Operations. A variant, the austere (or smaller) instrument group for solar observations, was also selected. In addition to the accessibility of mission data, (see Ref. 4.1), the scientists associated with the solar observatory criteria study were readily available for consultation.

4.2 MISSION OBJECTIVE AND EQUIPMENT

The scientific objectives of a large solar observatory (LSO) experiment are to study the physics of the photosphere and lower chromosphere, the upper chromosphere, and the corona. Six telescopes and associated instruments were selected to cover the wavelength range from the near infrared through X-rays. The solar observatory requirements for the 1.5-m (4.6-ft) telescope and instruments are shown in Table 4-1. This equipment list was reviewed for sortie applications by the Space Physics Laboratory at The Aerospace Corporation. This review suggested that the photoheliograph and the EUV spectrograph capability be extended to about 1000 Å. With such an extended capability in sensor technology, the ultraviolet telescope and spectroheliograph could be eliminated. The resulting LSO telescope and instrument list is summarized in Table 4-2. The geometric envelope of the instrument unit is shown in Figure 4-1. The data in Table 4-2 and Figure 4-1 were used in the design and cost analysis for both the LSO sortie and free flyer.

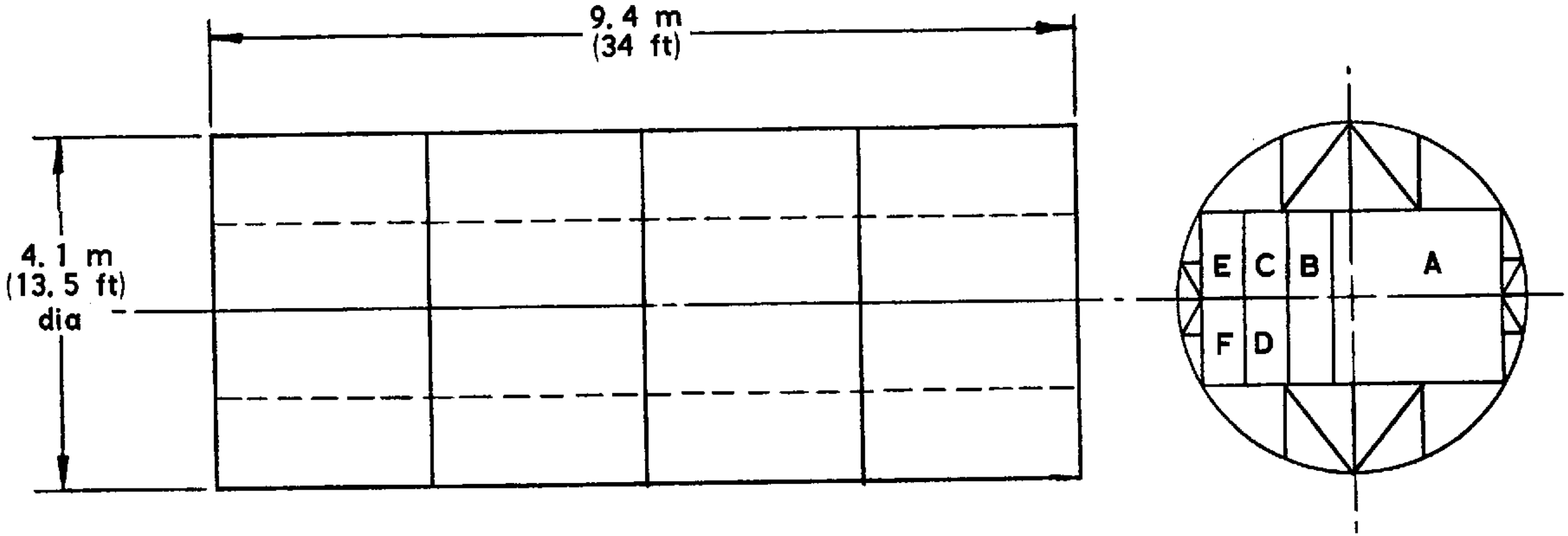
Table 4-1. Telescope and Instrument Requirements for the LSO

Instrument	Aperture m (ft)	Wavelength Range	Resolution arc sec	Field of View	Provisional Dimensions, m (ft)			Remarks
					Wt	Ht	Ln	
Photoheliograph	1.5 (4.6)	0.2-2.0 μm	0.1	4 arc min	1.8 (5.5)	2.0 (6.1)	3.0 (9.2)	Primary image photograph, spectroscopy
Spectrograph/ Spectroheliograph	--	0.2-1.5 μm	0.1	4 arc min	0.5 (1.5)	1.0 (3.05)	4.0 (12.2)	Spectra, magnetic and velocity fields
Ultraviolet Telescope	1.0 (3.05)	0.06-0.2 μm	0.1	4 arc min	1.3 (4.0)	1.5 (4.6)	3.0 (9.2)	Primary image photograph, spectroscopy
Ultraviolet Spectroheliograph	--	0.06-0.2 μm	0.1	4 arc min	0.5 (1.5)	0.5 (1.5)	3.0 (9.2)	Spectra, line profiles, photoelectron scan
Coronagraph	0.9 (2.7)	0.4-0.7 μm	1.0	15 deg	0.7 (2.1)	0.9 (2.7)	3.4 (10.4)	Corona, 1-6, 5-30 solar radii
Extreme Ultraviolet Spectroheliograph	0.6 (1.8)	100-600Å	0.5	5 arc min	0.5 (1.5)	1.0 (3.05)	10 (30.5)	Spectroheliograms, line profiles
X-Ray Spectroheliograph	0.6 (1.8)	5-140Å	1.0	5 arc min	0.5 (1.5)	1.0 (3.05)	10 (30.5)	Spectroheliograms, filtergrams,
Crystal Spectroheliograph	0.25 (0.76)	1-6 Å	1.0	1 arc sec (rastered)	0.5 (1.5)	2.0 (6.1)	8.0 (24.4)	Spectra line profiles
High-Energy X-Ray Collimator	0.5 (1.5)	0.3-1 MeV	1.0	full disk	0.5 (1.5)	1.0 (3.05)	6.0 (18.3)	Polarization bursts

Table 4-2. Telescope and Instrument List for LSO

Instruments	Weight kg (lb)	Power (W)	Data	Pointing/Stability* (arc sec)/arc sec/sec
Photoheliograph (2000Å - 20,000Å)	2,494 (5,500)	215	Film Video	(1)/0.02
Spectrograph/ Spectroheliograph	499 (1,100)	125	Video	(1)/0.02
Coronagraph (4000Å - 7000Å)	408 (900)	30	Film Video	(15)/0.5
EUV Spectroheliograph (100Å - 600Å)	454 (1,000)	125	Video	(2.5)/0.2
X-Ray Spectroheliograph (5Å-140Å) Filter	399 (880)	90	Video	(1.0)/0.1
X-Ray Spectroheliograph (1Å - 6Å) Crystal	385 (850)	80	Video	(2.0)/0.1

*Stability: Maximum Rate of Pointing



A = PHOTOHELIOGRAPH - 2 x 2 x 3 (meters) - 2,268 kg (5,000 lb)
B = CRYSTAL SPECTROHELIOGRAPH - 0.5 x 2 x 8 (meters) - 385 kg (850 lb)
C = X-RAY SPECTROHELIOGRAPH - 0.5 x 1 x 10 (meters) - 397 kg (875 lb)
D = CORONAGRAPH - 0.5 x 1 x 3.4 (meters) - 408 kg (900 lb)
E = SPECTROGRAPH/SPECTROHELIOGRAPH - 0.5 x 1 x 4 (meters) - 998 kg (2,200 lb)
F = EUV SPECTROHELIOGRAPH - 0.5 x 1 x 10 (meters) 454 kg (1,000 lb)

Figure 4-1. LSO Experiment Envelope - Full Complement

In addition to the LSO experiment list, an austere solar observatory (ASO) experiment set was also defined for this study. This list was based on the use of the 1-m (3.05-ft) photoheliograph being developed for the balloon-borne solar telescope. The 1-m (3.05-ft) unit could be supplied as flight-qualified hardware and any change would be limited to the focal plane, i.e., magnetograph and spectroheliograph. The remainder of the instruments were then sized to be consistent with the 1-m (3.05-ft) photoheliograph. This austere equipment list and characteristics are shown in Table 4-3. The packaged diagrams of these instruments are shown in Figures 4-2 and 4-3 for cylindrical and rectangular envelopes.

It should be recognized that the equipment data are preliminary and are still in the definition phase. The scientific community prefers to keep the definition open until the last milestone because of advancing technology and continuing research. Accordingly, the definition of the mission equipment is general and not in detail.

4.3 TRAFFIC MODEL

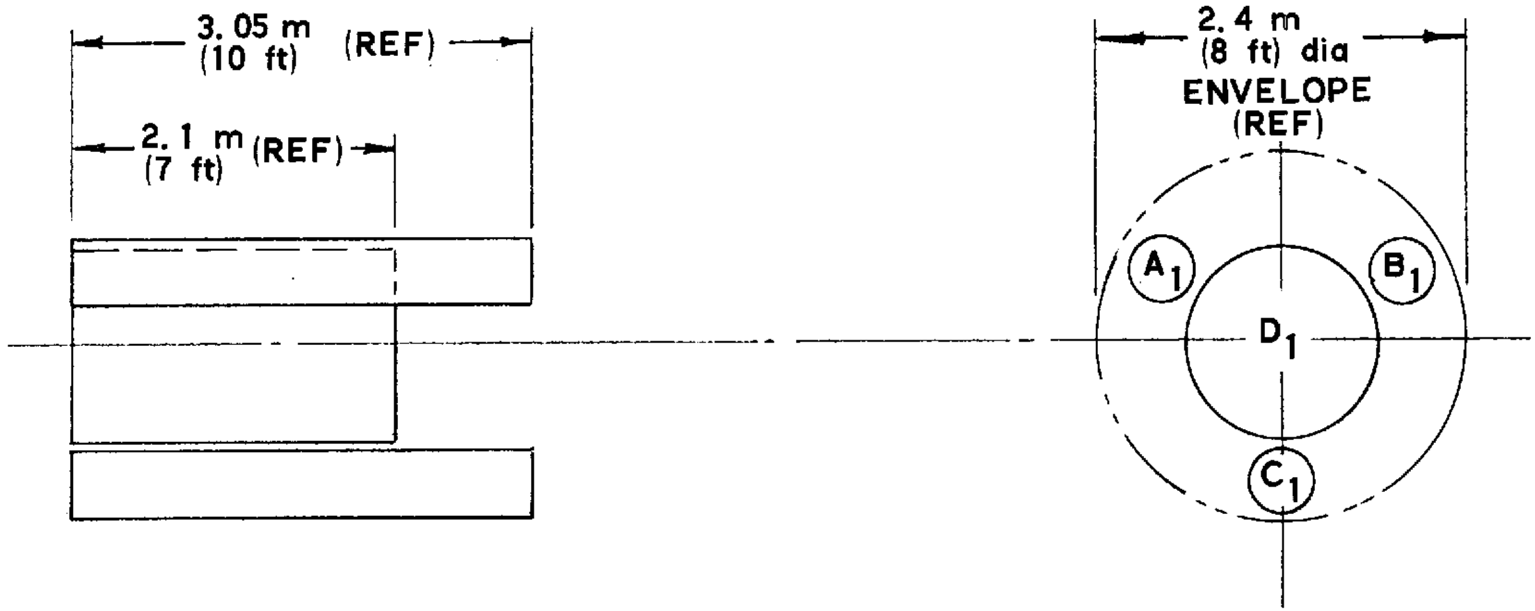
Sortie missions provide some characteristics that are not available with automated satellite and free flying observatories. They are more responsive to tailoring the flight to scientific needs and to updating mission instruments and equipments. The value of scientific data diminishes with mission duration because the equipment and instruments degrade with time, hence the higher priority objectives are scheduled early in the mission. Sortie missions can also be used to flight test sophisticated sensors before committing them to a large observatory payload as precursors to a large observatories program.

Evaluation of the various aspects of sorties requires their analysis in terms of economics. This was accomplished by developing four solar observatory programs which are judged to provide approximately equal

Table 4-3. Telescope and Instrument List for Austere Instrument Group - ASO

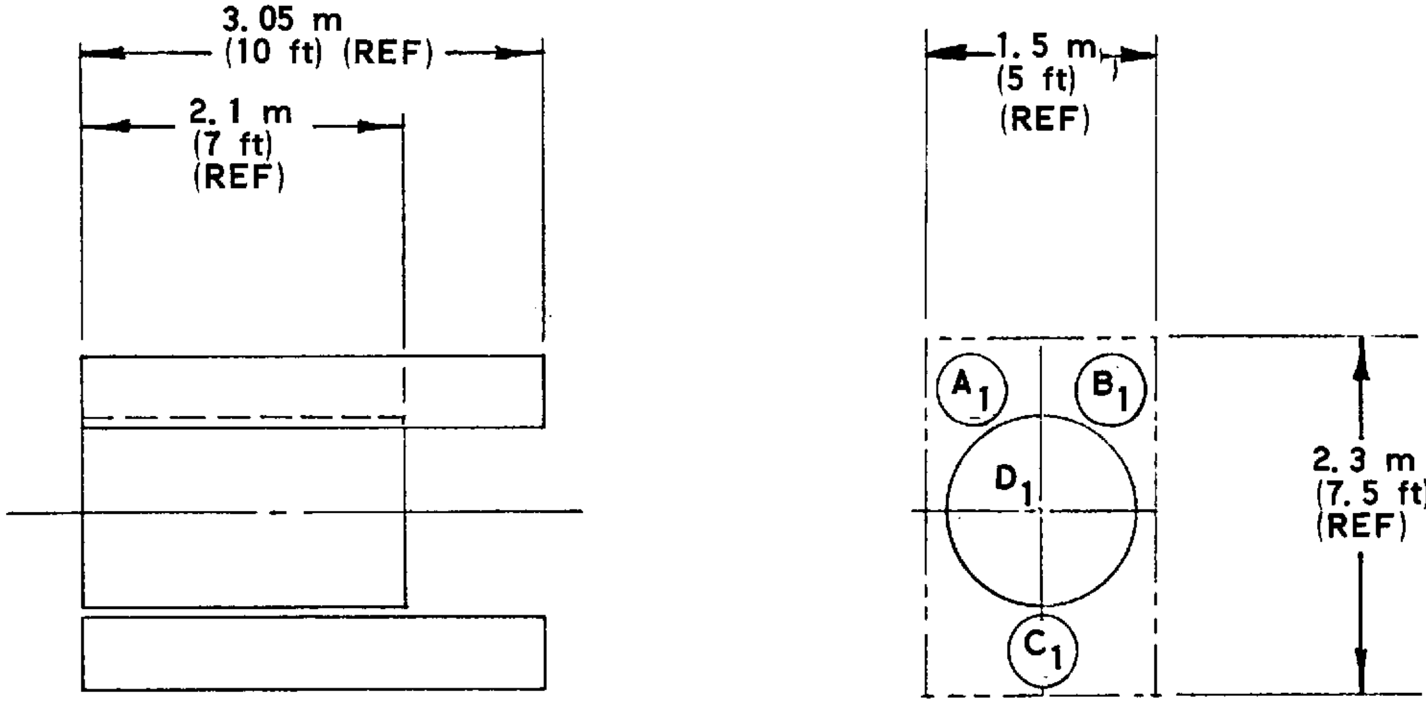
Instruments	Weight kg (lb)	Size			Power W	Data	Pointing/Stability sec/sec/per 1/2 orbit
		Wt m (ft)	Ht m (ft)	Ln m (ft)			
Photoheliograph (1 m)	1,360 (3,000)	1.2 (3.7)	1.3 (4.0)	2.1 (6.4)	120	Film Video	10/1
EUV Spectroheliograph	82 (180)	0.3 (0.9)	0.3 (0.9)	3.7 (11.3)	25	20 kbps	30*/5
X-Ray Spectroheliograph (Filter)	100 (220)	0.4 (1.2)	0.4 (1.2)	3.7 (11.3)	25	20 kbps	30*/5
X-Ray Spectroheliograph (Crystal)	77 (170)	0.4 (1.2)	0.4 (1.2)	3.7 (11.3)	25	20 kbps	30*/5

*Fine pointing control by experimenter.



A_1	= EUV SPECTROHELIOGRAPH	43 x 43 x 305 cm (17 x 17 x 120 in.)
B_1	= X-RAY SPECTROHELIOGRAPH (FILTER)	43 x 43 x 305 cm (17 x 17 x 120 in.)
C_1	= X-RAY SPECTROHELIOGRAPH (CRYSTAL)	43 x 43 x 305 cm (17 x 17 x 120 in.)
D_1	= 1 METER PHOTOHELIOGRAPH	122 x 135 x 213 cm (48 x 53 x 84 in.)

Figure 4-2. Austere Instrument Group, Orbiter/ASO



A_1 = EUV SPECTROHELIOGRAPH	43 x 43 x 305 cm (17 x 17 x 120 in.)
B_1 = X-RAY SPECTROHELIOGRAPH (FILTER)	43 x 43 x 305 cm (17 x 17 x 120 in.)
C_1 = X-RAY SPECTROHELIOGRAPH (CRYSTAL)	43 x 43 x 305 cm (17 x 17 x 120 in.)
D_1 = 1 METER PHOTOHELIOGRAPH	122 x 135 x 213 cm (48 x 53 x 84 in.)

Figure 4-3. General Purpose Lab Austere Instrument Group - Orbiter/ASO

value in scientific data return and coverage. The programs formed in this manner are shown in Table 4-4. The rationale for the programs is as follows:

I. Sortie

The sortie is augmented by the automated satellite which will provide the continuous coverage. The automated satellite is of the Orbiting Solar Observatory (OSO) class. The sortie payload is the LSO and consists of the free flyer equipment. During the 11-year solar cycle, there are two scheduled and one unscheduled sortie flights. The unscheduled sorties are those solar experiments that are to be performed with only 13-day notice. If the automated satellite were positioned in superior conjunction orbit, it could detect data for monitoring by sortie 1/2 sun rotation before it comes in earth view, the sun rotation period being 26 days. Two flights are scheduled at maximum sun spot activity periods and one at minimum activity period.

II. Sortie and Free Flyer

This program is included to accommodate the case where the sortie flights are used as precursors to the free flyer. The same LSO mission equipment is used for both payloads. Sortie will be the test bed in the development of the sensors and will also perform early experiments. Supplementing the scheduled sortie missions, the unscheduled sortie flights and automated satellites (OSO) are included to provide the continuous coverage. The free flyer is considered to be the National Solar Observatory and is independent of the Shuttle or space station during the orbital operations; it depends, however, on the Shuttle for the yearly maintenance, repair, and resupply visits.

Table 4-4. Traffic Models for NASA Solar Observatory Program

No.	Program Approach	Type Payload	Schedule, Year											
			79	80	81	82	83	84	85	86	87	88	89	90
I	LSO Sortie	Scheduled sortie	2	2	2	1	1	1	1	1	1	1	1	2
		Unscheduled sortie	1	1					1					
		Automated P/L (OSO)*		1		1		1		1		1		1
II	LSO Sortie + Free Flyer	Scheduled sortie	2	2	2									
		Unscheduled sortie	1	1										
		Free flyer				1								
		Manned visits					1	1	1	1	1	1	1	2
		Automated P/L (OSO)		1		1								
III	Free flyer	Free flyer	1											
		Manned visits	1	2	2	1	1	1	1	1	1	1	1	2
IV	ASE Sortie + Free Flyer	Scheduled sortie	2	2	2	1	1	1	1	1	1			
		Unscheduled sortie	1	1										
		Free flyer								1				
		Manned visits									1	1	1	2
		Automated P/L (OSO)		1		1		1		1				

*orbiting solar observatory in operation since 1962

III. Free Flyer

This concept, esentially the program described above, is included as the baseline concept. In this plan the free flyer has an early start date and does not obtain any technology development data from sortie missions. The schedule shows that visits are increased during the high solar activity periods.

IV. ASO Sortie and Free Flyer

The experiment packages for the sortie are austere in comparison to the free flyer. Basically there will be two equipment development periods, since the sortie will use the ASO instruments. The free flyer, however, will use LSO instruments and the technology from the sortie. This concept should result in maximizing the capability of the free flyer observatory because of the sortie development flights. The free flyer will gain from the sortie experience.

4.4 DESIGN

The design study of the sortie/solar observatory was conducted to conceptually examine various sortie approaches which were intended to include the major factors influencing its design. These factors are manned vs unmanned operations, low vs high pointing accuracy, small vs large sensor aperture, LSO vs austere sets of mission equipments, and sortie vs free flyer. The conceptual design effort was undertaken to provide methods which would trade off the postulated influencing factors. These conditions resulted in the sortie approaches shown in Table 4-5.

Table 4-5. Sortie Configuration (A → H), Summary of Mission Equipment Support Features

LSO Mission Equipment Support Features	ASO Mission Equipment Support Features
A - Unmanned Operations Hardmate CMG added to orbiter	E - Unmanned Operations Gimbal/Torquers
B - Unmanned Operations Gimbal/Torquers	F - Manned Operations Gimbal/Torquers Dedicated Lab
C - Unmanned Operations Tethered	G - Manned Operations Gimbal/Torquers Share lab facility
D - Unmanned Operations Free flyer*	H - Manned Operations Gimbal/Torquers Attach to lab

*Free flyer is not considered a sortie.

Manned vs unmanned operations are examined in the austere set by comparing configuration A vs B, C, or D. The scientists have a strong preference for manned operations. The term "manned" is used to denote the capability of man to repair and adjust the instruments in orbit, between mission observation phases. During mission observation phases, the instrument bay is vacated for depressurization to permit sensor operation and contamination control. The term "unmanned" implies that the mission instruments are adjusted remotely. In both cases two mission specialists are available to monitor the mission operation and data.

The level of pointing accuracy is examined in the LSO set by improving the pointing capability from A through D. The free flyer included in the LSO, which is not considered a sortie, is included in this analysis only for comparison. The free flyer should provide the highest pointing accuracy because of its physical separation from the orbiter.

4.4.1 LSO Design Concepts

The LSO experiment complement as defined in Table 4-2 occupies an envelope approximately 4.1 m (13.5 ft) in diameter and 10.4 m (34 ft) long (see Figure 4-1) which was developed from the geometric data provided in Table 4-1. This geometry permits identification of four configurations that can integrate the full LSO experiment package into the orbiter with varying degrees of pointing accuracy. These configurations range from a hard-mounted deployable LSO system (configuration A) to an autonomous free flyer (configuration D). These concepts, illustrated in Figures 4-4 through 4-7, are controlled remotely (unmanned).

Configuration A, Figure 4-4, shows the experiment package mounted on the orbiter-supplied deployment device and rotated 90 deg out of the cargo bay. The LSO is rigidly attached to the Shuttle and the entire combination is initially aimed at the sun by the Shuttle's attitude control

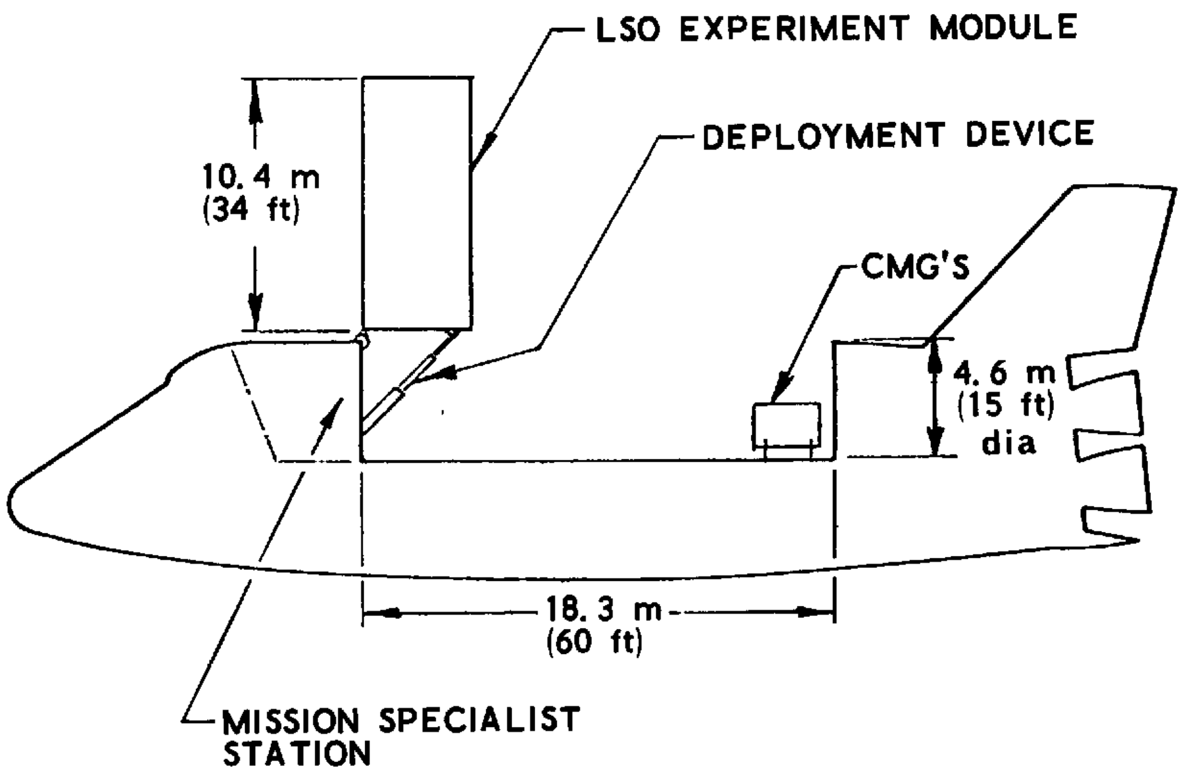


Figure 4-4. Configuration A, LSO - Hard-Mate, Deployed, and Remote Control

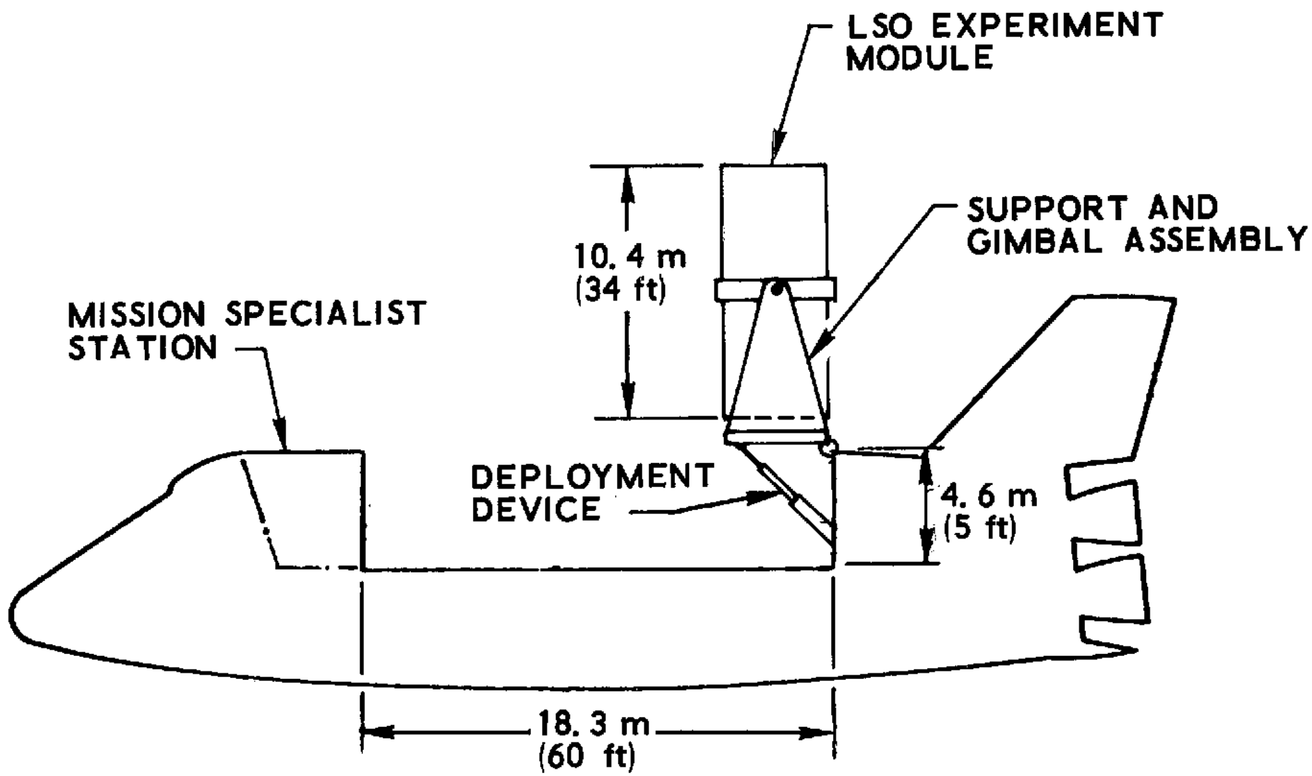


Figure 4-5. Configuration B, LSO, Hard-Mate, Gimballed

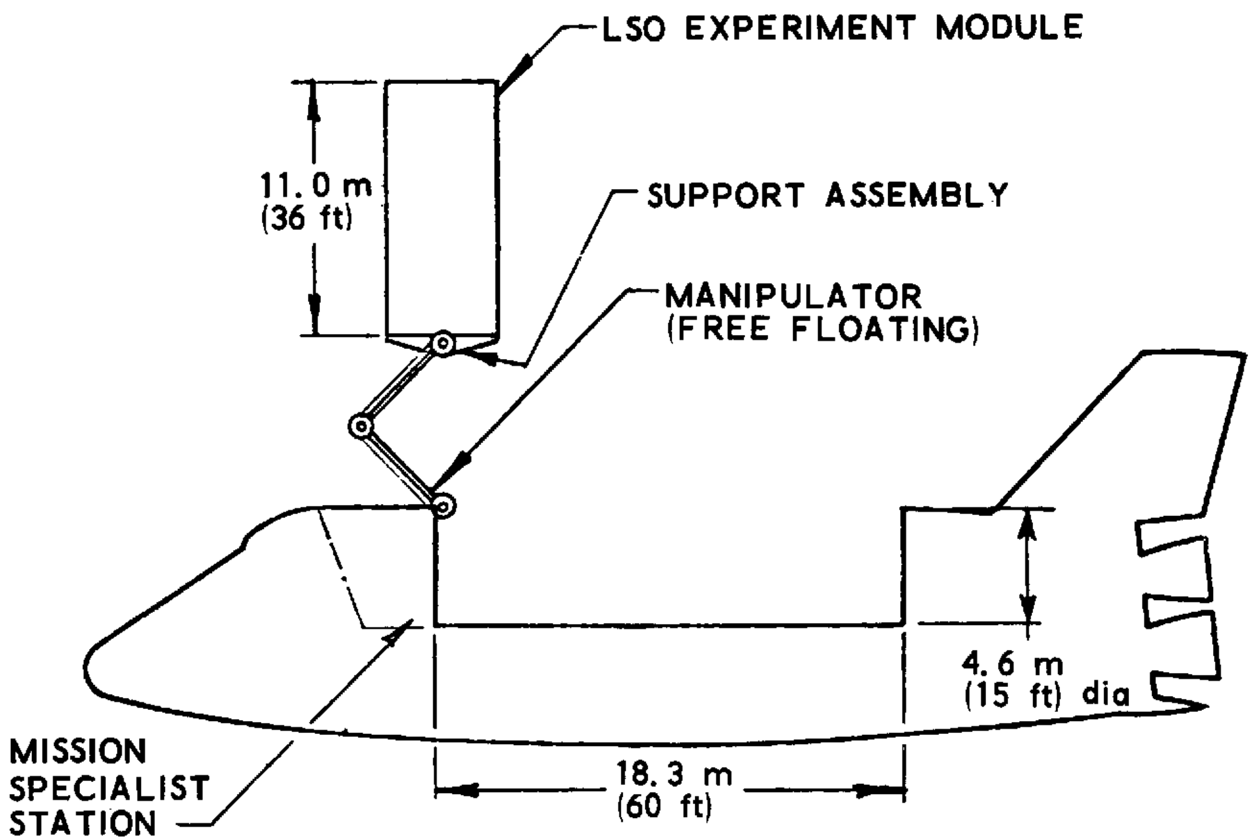


Figure 4-6. Configuration C, LSO - Tethered Manipulator and Free Floating

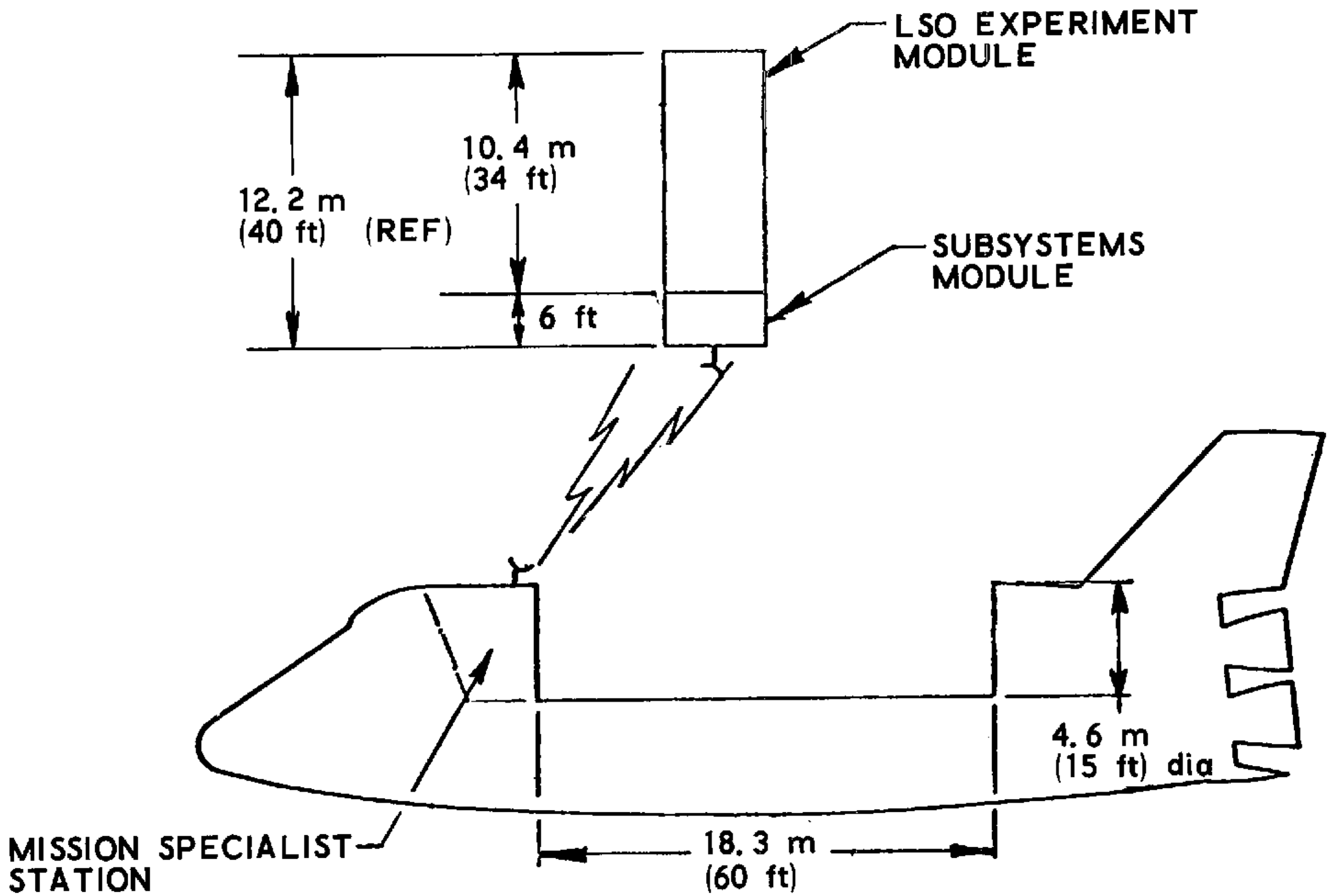


Figure 4-7. Free Flyer Mission, Large Solar Observatory - Configuration D, LSO, Telemetry Control

system. This concept postulates the addition of control moment gyros (CMGs) to augment the orbiter and to control the LSO and orbiter as a single unit for the experiment pointing capability. The final pointing accuracy is maintained by the CMG in the orbiter payload bay.

Configuration B, Figure 4-5, is also deployed out of the cargo bay; but in this concept standard orbiter attitude control is employed, torquers and three axis gimbals being added to the LSO to provide experiment pointing. The LSO is mounted on a three axis gimbal which provides the fine pointing with the orbiter attitude control system merely maintaining the Shuttle original position within $\pm$ 0.5 deg.

Configuration C, Figure 4-6, makes use of the orbiter-furnished remote manipulator to deploy and position the experiment assembly. When this is accomplished the torquers on the gimbals in the manipulator arms may be released and the LSO can maintain its position with its own attitude control system and momentum wheels, or possibly the manipulator gimbal torquers can be utilized by the LSO for attitude control. Power and communication, however, are supplied from the orbiter (as in configurations A and B) through a cable attached to the manipulator.

Configuration D, Figure 4-7, is a free flyer operating under radio control in the vicinity of the orbiter. It is positioned and retrieved with the orbiter remote manipulators as in the other configurations. The free flyer and orbiter must station-keep during the orbital operation. The free flyer contains its own subsystems support module; i.e., electrical power, communication, environmental control subsystems, attitude control, and reaction wheels.

Study results summarizing the estimated pointing accuracy for configurations A through D are listed in Table 4-6 and are discussed in section 4.4.4. It is clear that concepts A and B cannot satisfy the indicated pointing requirements of Table 4-2. It is generally agreed that configurations C and D can meet the postulated requirements if some form of image motion compensation is included in the experiment instrumentation.

It should be recognized that the free flyer (configuration D) is not considered a sortie. It is apparent, however, that to achieve the pointing requirements the mission equipment must approach free flyer condition.

4.4.3 Austere (ASO) Design Concepts

Four feasible installation configurations were identified for the ASO mission equipments. Two versions of the ASO payload are installed in the orbiter cargo bay, one with man access and the second without. Two concepts involve the use of the postulated General Purpose Laboratory (GPL) with man access capability. The ASO concepts are all gimballed/torque controlled for pointing accuracy.

Geometric data describing the ASO instrument complement and experiment group envelope are given in Figure 4-2. This envelope was used for the ASO experiment group installed in the orbiter cargo bay. The same experiments and instrument sizes were used in all installation concepts studied; a slight change in experiment group envelope size and shape, however, was used for the experiments installed in the GPL in order to better suit the diameter of the assumed GPL -- 4.3-m (14-ft) outside diameter compared with a permissible 4.6-m (15-ft) diameter cargo bay payload envelope. The experiment envelope for the GPL configuration is shown in Figure 4-3.

Table 4-6. Sortie Solar Observatory, Control Approaches

Configuration	Control Concept	Remarks
A - Hardmate Deployed	Control moment gyros in orbiter	CMG unit hard mounted to orbiter LSO and orbiter attitude control as single unit 5 arc sec pointing accuracy 10 arc sec/sec stability*
B,E, F,G, H Gimballed (manipulator to deploy only)	Experiment package gimballed	Standard orbiter attitude control Gimbal support assembly locked with respect to orbiter Payload moves with respect to orbiter Gimbal drives torque against orbiter 2 arc sec pointing accuracy 5 arc sec/sec stability*
C - Manipulator Tethered	Reaction wheels in experiment package	Standard orbiter attitude control Manipulator actuators free to rotate Reaction wheels mounted in LSO No control reaction torque on orbiter - only geometrical constraint torque 1.0 arc sec pointing 3 arc sec/sec stability*
D - Free flyer (not a sortie)	Reaction wheels in observatory	Standard orbiter attitude control Reaction wheels in LSO No physical connection between LSO and orbiter 0.1 arc sec pointing accuracy 1 arc sec/sec stability*

*without image motion compensation

Configuration E, Figure 4-8, illustrates the unmanned version of the orbiter-supported ASO payload. In this approach the instrument must be adjusted remotely. (The installation concept is shown in some detail in the larger-scale drawing of Figure 4-12.) The experiments are installed within a 2.4-m (8-ft) diameter shell structure sized in conformance with the circular envelope of Figure 4-2. The instruments are attached to each other and are supported within the instrument shell structure by means of an appropriate internal truss. The inside diameter of the shell structure supports a heat-pipe assembly (see section 4.4.5) that is used to provide a uniform temperature atmosphere and minimize the longitudinal and lateral thermal gradients within the instrument group and instrument support structure. A fluid coolant loop and radiator are provided to maintain the operating temperature of the experiment group to ambient room temperature. Three axis stabilization of the experiment package is provided by azimuth, elevation, and roll gimbals provided on the experiment support structure. Retractable launch locks are provided between the payload and the standardized payload pallet that serves to secure the payload and to transfer launch loads from the payload support structure to the cargo bay hard points.

Configuration F, Figure 4-9, illustrates the ASO experiment grouping used in a concept that can permit manned access to the payload for repair or adjustment of the sensors. This feature would require the use of a flexible tunnel between the orbiter mission specialist station and the entrance of the experiment canister. In this concept the experiment canister and flexible tunnel must be pressure tight. Sufficient atmosphere, chargeable to the payload, is provided to allow a total of nine man-hours of access to the experiments in three pressurizations. The instrument canister is evacuated when experiments are in process. The experiment is deployed out of the cargo bay to improve the performance of the experiment radiator loop.

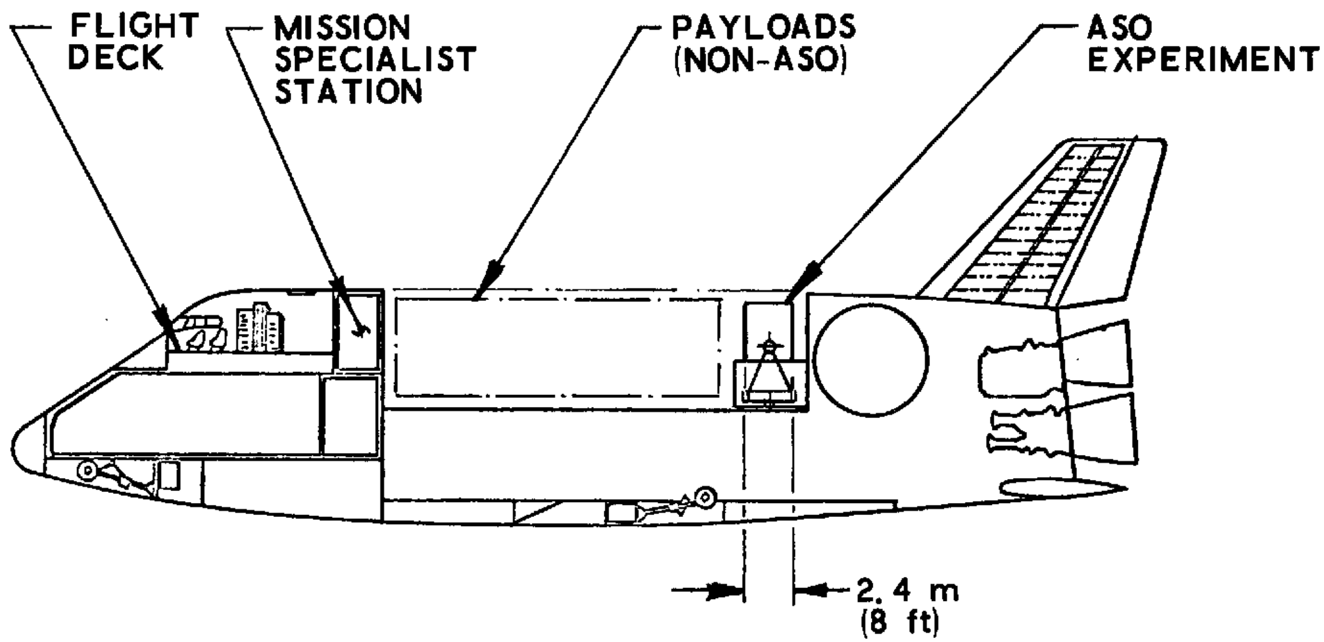


Figure 4-8. Configuration E, Austere - Gimballed, Non-Deployed ASO, and Unmanned

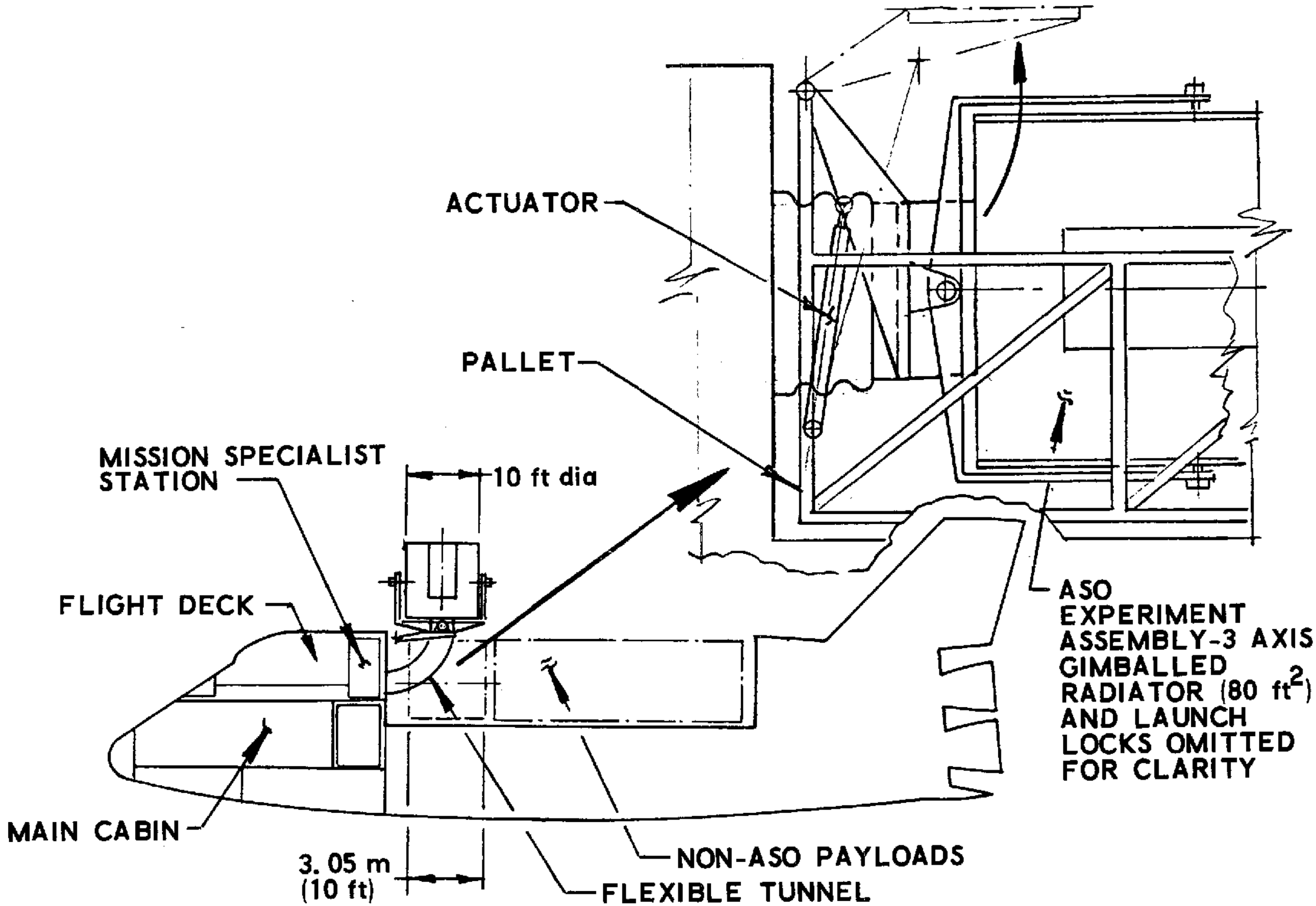


Figure 4-9. Configuration F, Austere - Gimballed, Deployed, and Manned ASO

Configurations G and H, Figures 4-10 and 4-11, illustrate two concepts for installation of the ASO in association with the GPL. The concept is shown in some detail in Figure 4-13. The basic features of the GPL ASO concept are similar to those described above for the manned ASO except that some form of rotating or sliding door must be provided to permit pressurization of the ASO equipment compartment. A pressure bulkhead is provided in the Figure 4-10 concept to isolate the ASO compartment from the remainder of the GPL. In this concept it is assumed that there is sufficient volume within the GPL to accommodate the ASO compartment and the experiment equipment associated with a number of other scientific disciplines, and that some subsystems (e.g., ECLS) could be shared among the various on-board disciplines. The Figure 4-11 concept provides a separate and dedicated ASO module that would provide its own subsystems support. The Figures 4-10 and 4-11 configurations are shown deployed in phantom lines to indicate that some disciplines may require deployment. If this were to be the case, the data-gathering time available to a particular scientific experiment discipline might be reduced and the total on-orbit time would have to be shared.

These pressurization modules are shown with ASO experiments; they can be replaced by other experiments, however, such as astronomy, space physics, and earth observation. This would permit high usage and cost sharing of the basic module.

4.4.4 Stabilization and Control

Stabilization and control were briefly examined to determine the various approaches and to estimate their capabilities. As stated above, the mission equipment and orbiter are not sufficiently defined to perform detail analyses; weight and pointing accuracies, however, were estimated from the type of controls for the various concepts. (See Table 4-6.) Thrusters were not included because of the potential contamination

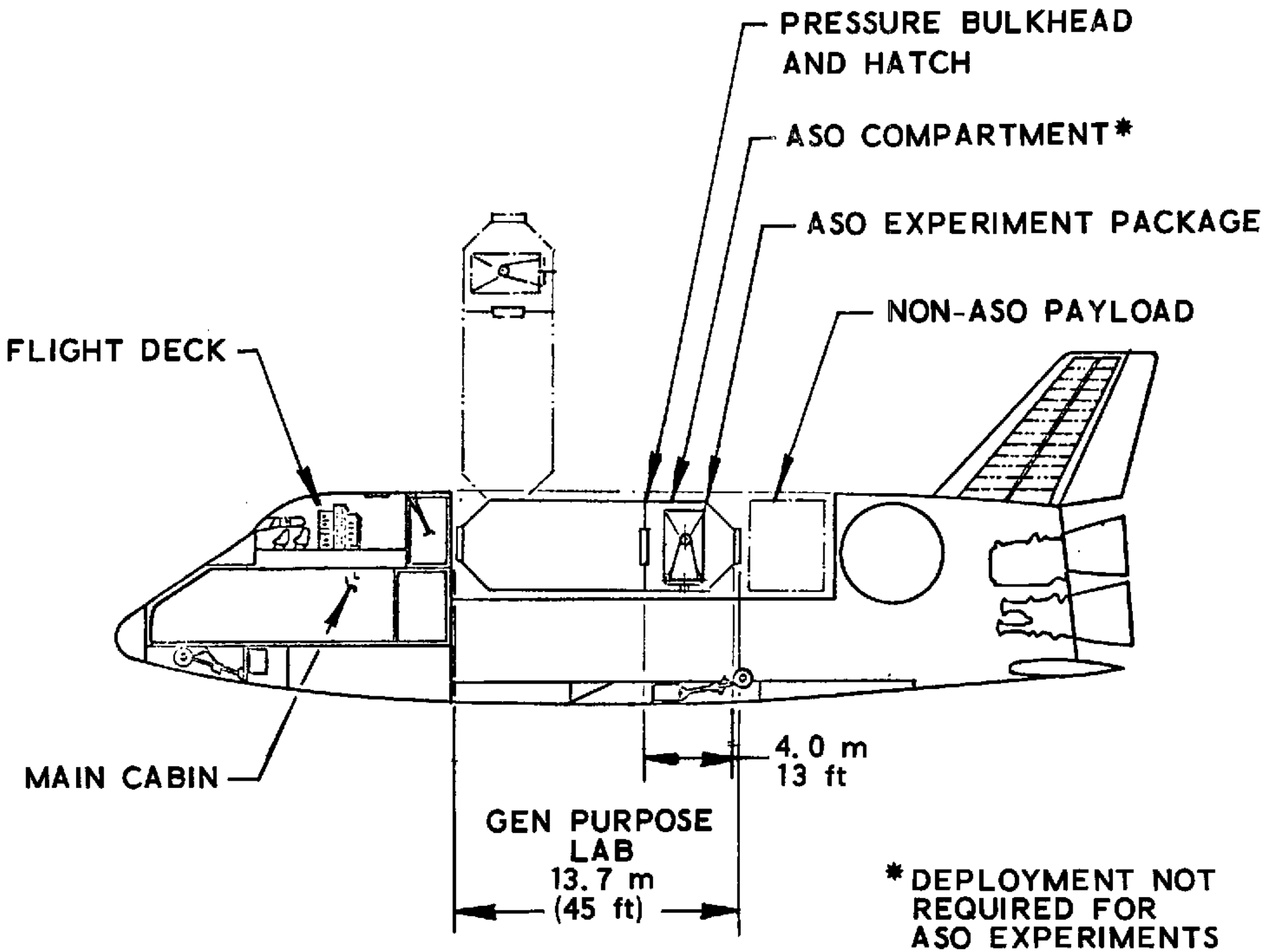


Figure 4-10. Configuration G, Austere - Gimballed, Deployable Shared GPL and Manned

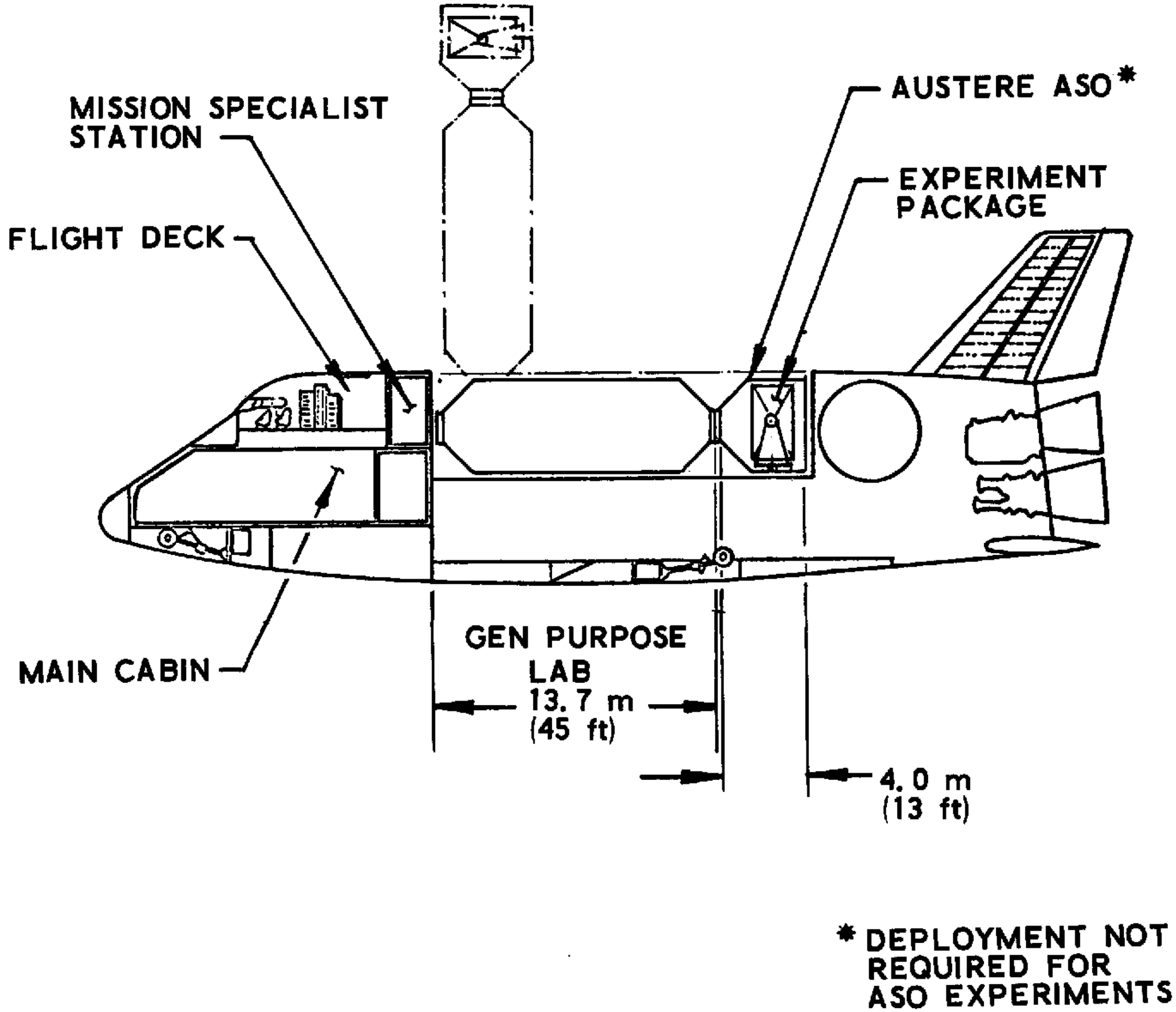


Figure 4-11. Configuration H, Austere - Gimballed, Deployable Tandem GPL, and Manned ASO

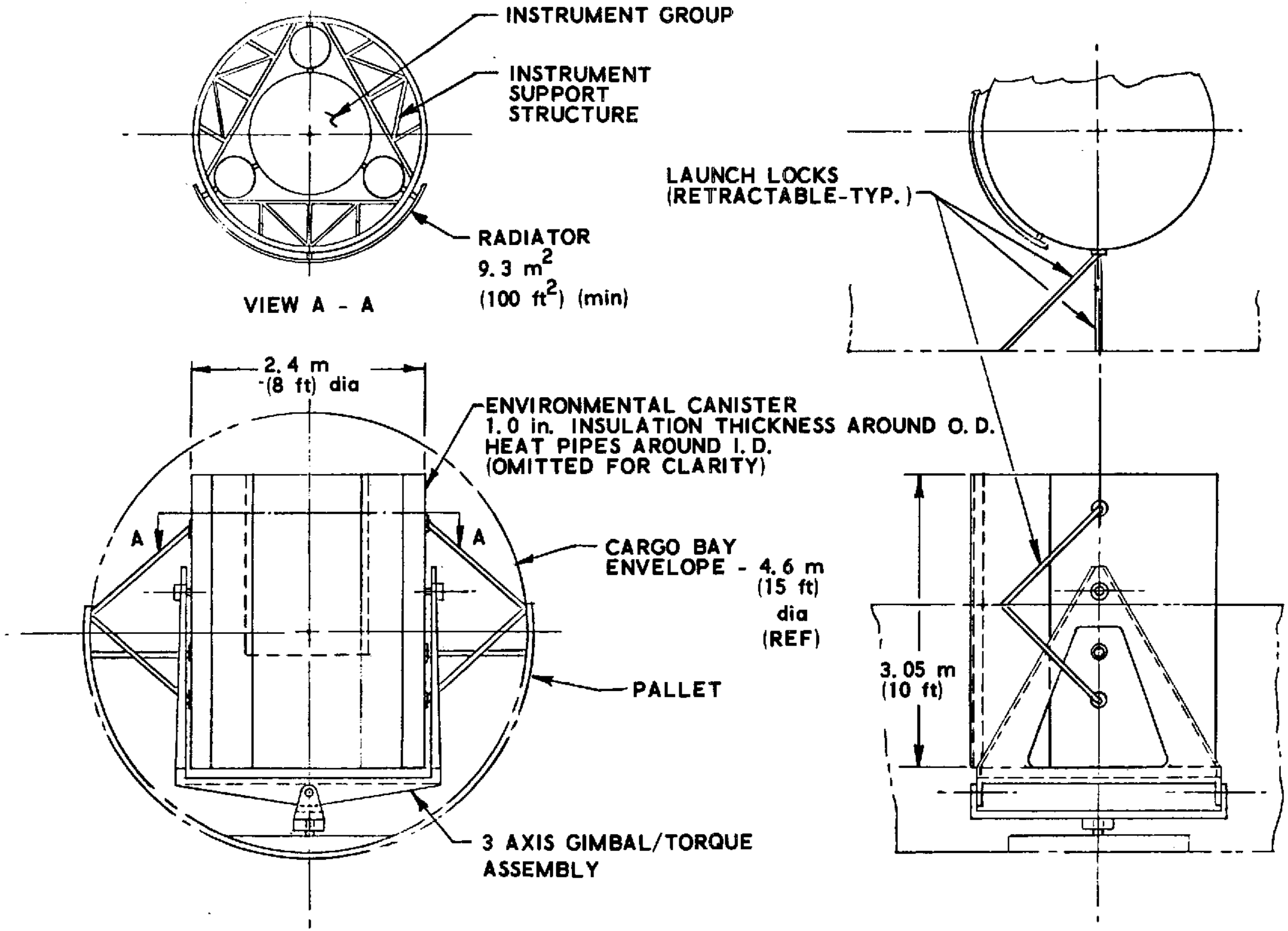


Figure 4-12. Orbiter/ASO Installation - ~~Unmanned~~ Experiment, Configuration E

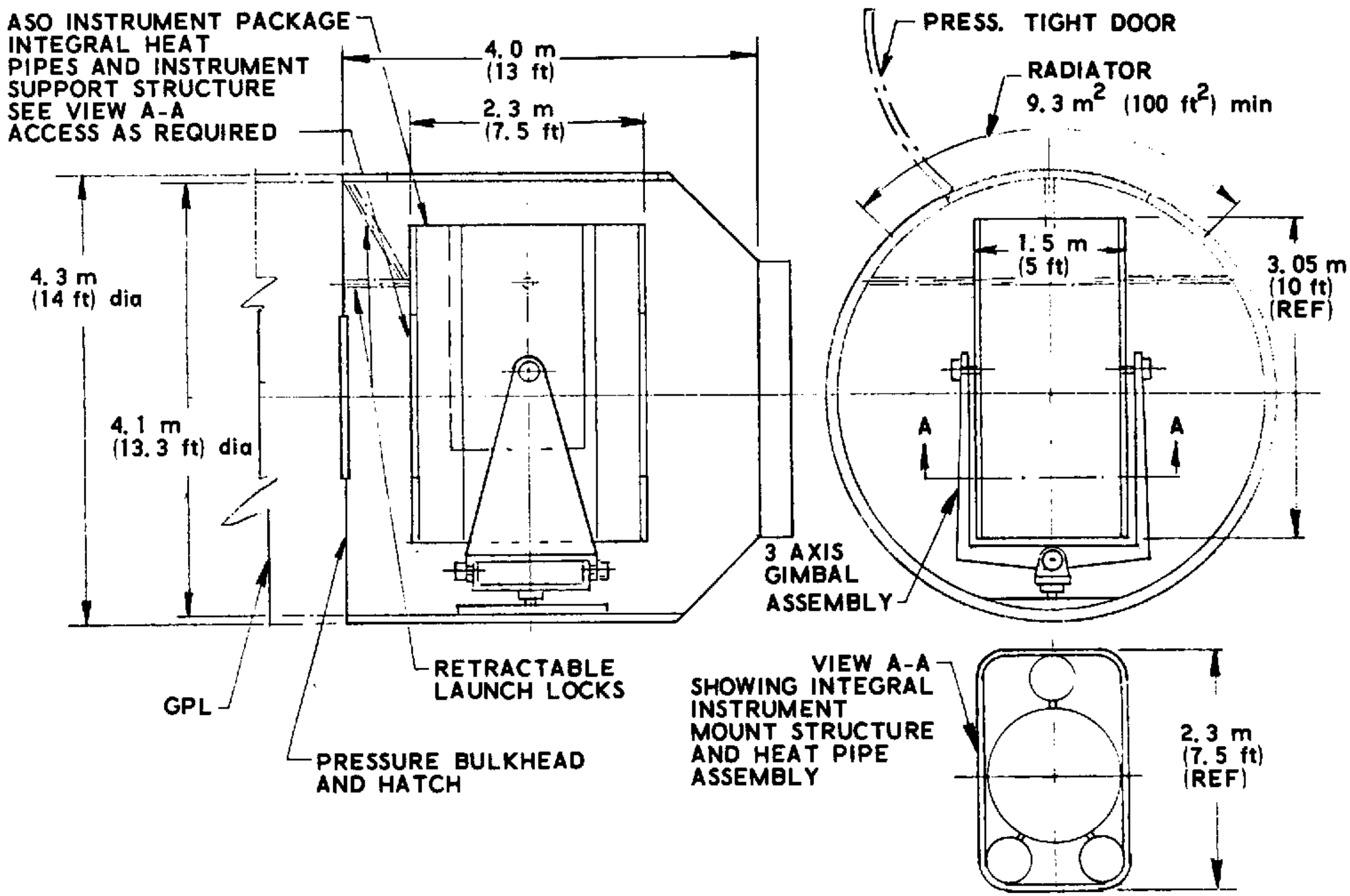


Figure 4-13. ASO Experiment Installation - GPL, Configuration H

problems. The optic performances are sensitive to contamination. Lens covers would probably be required during those phases when the Shuttle thrusters are in operation.

The orbiter stabilization capability assumed in this study was $\pm$ 0.5 deg continuous pointing accuracy for one orbit every other orbit (Ref. 4.1). The mission equipment pointing requirement is 1 arc sec for the LSO version and 10 arc sec for the ASO version (see Tables 4-2 and 4-3). Stability augmentation is therefore required if the mission requirement is to be met. The basic stability approaches are to augment the orbiter pointing, or to separate the mission equipment and augment only the equipment. Image motion compensation (IMC) or other internal instrument alignment compensation techniques were not considered in this study, but may be incorporated into the payload as required.

For the orbiter stability augmentation the mission equipment can be hardmated to the orbiter, a tactic which should simplify the design installation and operation. A CMG would provide the fine torquing, but would be limited in its pointing capability because of internal disturbances including crew motion, rotary machinery, and large extendables. If these disturbances can be minimized, however, and if the sensors are located on the experiment platform and the overall orbiter/payload is stiff, it is expected that five to ten arc sec pointing is feasible. For this study three double gimbal CMGs were selected with an angular momentum of approximately 1,383 m-kg (10,000 ft-lb)/sec. A CMG of that size should provide roughly 0.1 deg/sec maneuvering rate and weigh approximately 363 kg (800 lb) each (Ref. 4.3). The total CMG package weight including structure, electronics, sensors, and cables is estimated at 1,587 kg (3,500 lb). This size CMG unit was considered for the LSO and would be used for the ASO since the mass of the mission equipment is small in comparison to the orbiter.

The second approach is to gimbal the mission equipment to increase the pointing accuracy. The equipment will have sun sensors which are mounted on the experiment platform to drive the azimuth and elevation gimbals. The roll axis about the line of sight must be controlled to maintain the orientation about the orbiter. In this case the star sensor, instrument measurement unit (IMU), and computer in the Shuttle will be used for the gimbal transformation necessary to drive the appropriate gimbals for the positioning about the orbiter. Gyros will probably be needed in the loop for low jitter requirements. In this concept the torquer characteristics for the LSO and ASO sorties are shown in Table 4-7 for 0.01 rad/sec^2 response characteristics. The ASO gimbal size is substantially less than the LSO gimbal because of the smaller equipment package and mass. The gimbal structure to provide rigidity in transferring the torques to the orbiter was based on contractor studies (Ref. 4.4).

The third stabilization approach is the tethered LSO free flyer concept. The tether is considered to be the manipulator in a free mode. The attitude control system will be basically a free flying approach with the manipulator to provide the deployment services and initial pointing. The LSO is not physically freed because electrical power and communication functions are supplied by the orbiter via an umbilical played alongside the manipulators. It can be expected that the tether will introduce some disturbances, since completely free joints are not physically possible. The attractive feature of the free floating manipulator, besides providing the deployment mechanism and the link for subsystem functions, is that it can be used as a mechanical station-keeping device. In the event that either orbiter or payload control malfunction or drift occurs, the manipulator can be locked to prevent collision.

Table 4-7. Stabilization Subsystem Characteristics

Characteristics	Sortie Mode		
	Units	LSO	ASO
Mission Equipment Mass	kg (lb)	4,910 (10,825)	1,814 (4,000)
Azimuth and Elevation Torquers	kg-m (ft lb)	124 (900)	5 (35)
Roll Torquers	kg-m (ft lb)	28 (200)	5 (35)
Average Power	W	400	80
Total Weight	kg (lb)	728 (1,605)	93 (205)

The complete free flyer approach is introduced to provide a baseline for comparison. Furthermore, the free flyer is the goal of the National Solar Observatory and should be compared with it. The free flyer is not considered a sortie even though the orbiter is station-keeping in close proximity. A mechanical tie is required between the orbiter and the payload to permit sortie classification.

4.4.5 Environmental Control and Life Support (ECLS)

Thermal and consumables analyses were briefly conducted to determine the type and weight of the ECLS. This analysis included the following assumptions:

(1) 555-km (300-nmi) sun-synchronous twilight orbit

(2) Solar observatory pointed towards the sun when operating

(3) Ambient (room) temperature for nominal operating conditions

(4) Sensor thermal loads considered black box loads

(5) Manned occupancy for instrument adjustments and repairs; three visits per sortie mission for three man-hours each, but limited to two hours total elapsed time per access

(6) Thermal gradient to be limited to $\pm$ 2.8°C ($\pm$ 5°F) transversely and $\pm$ 1.4°C ($\pm$ 2.5°F) axially

(7) Thermal loads as follows:

Thermal Loads	ASO 1 m (3.05 ft)	LSO 1.5 m (4.8 ft)
Primary Solar Load	200 W	500 W
Secondary Solar Load and Electronics	700 W	3100 W
Thermal Leakage	30 W	100 W
Pump	70 W	300 W
Total Energy	1000 W	4000 W

4.4.5.1 Mission Equipment Thermal Control

To meet the requirements it was determined that an active cooling loop is required to maintain the temperature gradient and to transfer the solar load from the optics, and that heat pipes and thermal insulation are required for the structure supporting the secondary optics. Thermal studies were performed by Itek Corporation (Ref. 4.2) which resulted in similar findings.

The cooling system provides the necessary heat transport medium for the optics to take thermal loads from the sensor thermal interface to the radiator. The coolant is an 80/20 methanol, which is the same fluid as that used in the Apollo Telescope Mount (ATM) cooling system. The minimum required flow rate is 268 kg/hr (590 lb/hr) heat exchanger; the system can operate, however, at an ATM pump nominal flow rate of approximately 408 kg/hr (900 lb/hr).

A diagram of the coolant loop is shown in Figure 4-14 for undeployed and deployed equipment. The basic components in the coolant loop are the sensor interface cold plate, fluid accumulator, pump, radiator, radiator bypass line, and heater. There are three cold plates. The first cold plate is for the 200-W solar load; the second for the 20 to 30-W heat pipe load, which intercepts heat leakage through the insulation; and the third for the 700-W solar and mission equipment power load. Fluid temperature into the first cold plate is controlled at 28° C $\pm$ 0.8° C (82° F $\pm$ 1.5° F) for the undeployed mode. The $\pm$ 0.8° C (1.5° F) tolerance on the coolant inlet temperature is consistent with prior coolant system studies that were similar to this approach. The rather high temperature is dictated by the poor radiator view to space factor and to the sides of the cargo bay. The radiator that is mounted on the side of the LSO has a view factor of unity with the side walls of the cargo bay (see Figure 4-15). The cargo bay was assumed to be painted with an α/ϵ = 0.25 paint which results in a bay temperature of 7° C (45° F).

The resulting radiator area required is approximately 15.8 m^2 (170 ft^2) with fluid temperature at 28° C (82° F) for the ASO undeployed case. For the deployed mode (view factor to space $\approx$ 0.5), the radiator area can be reduced to approximately 4.6 m^2 (50 ft^2) and the fluid temperature into the first cold plate can be held to 21° C $\pm$ 0.8° C (70° $\pm$ 1.5° F). Total fluid temperature rise, based on a flow rate of 408 kg/hr (900 lb/hr) and 925 W will be approximately 2.8° C (5.2° F).

Cooling system weight for the ASO unmanned deployed and undeployed cases are 38.5 and 136.1 kg (85 and 300 lb), respectively. The cooling system weight does not include meteoroid protection weight. For the manned modes, an additional 6.8 kg (15 lb) is added to account for a heat exchanger to remove human heat loads. The power necessary to operate this system is 65 W of continuous power for the pump and up to 200 W of intermittent power for the heaters.

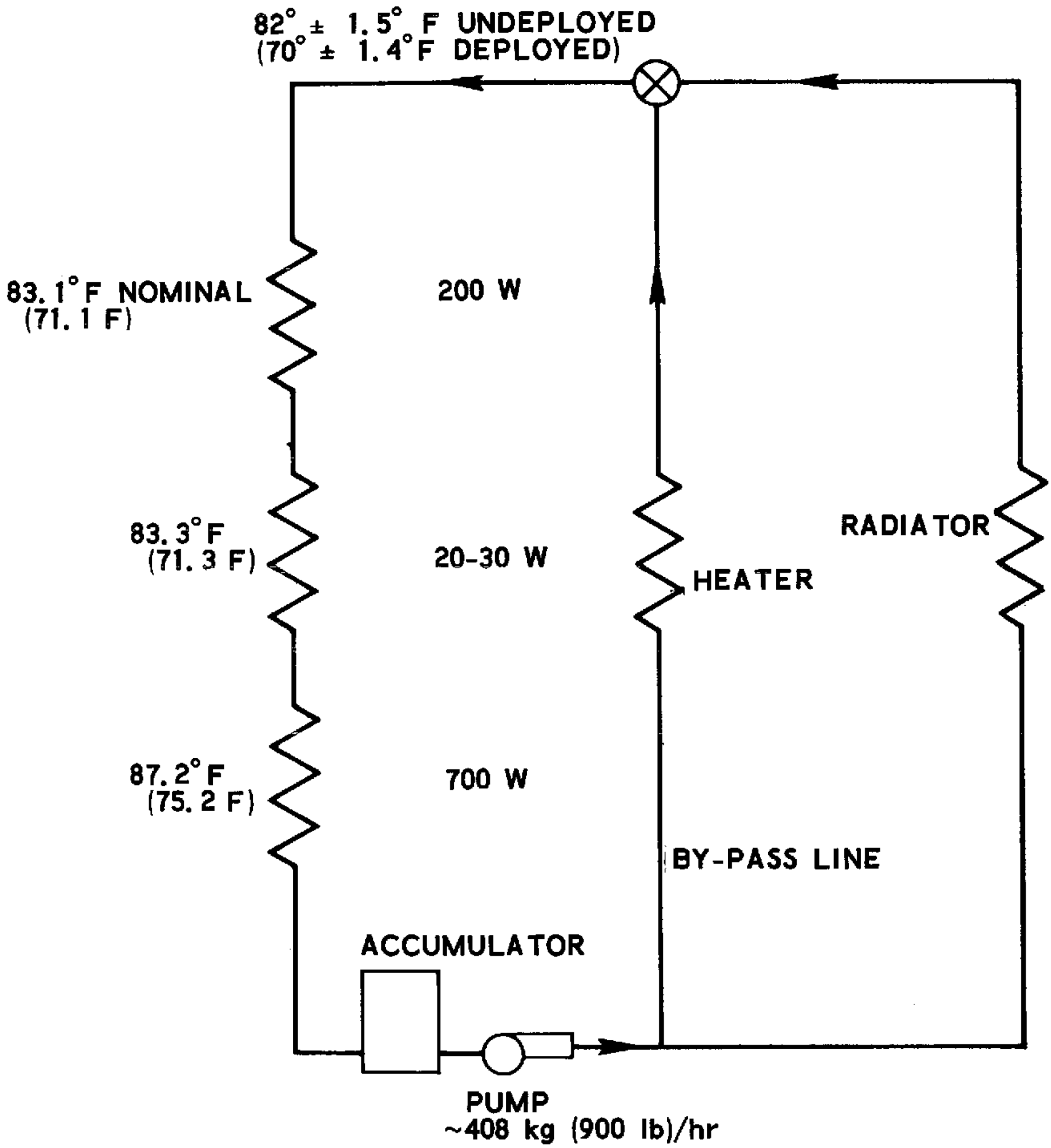


Figure 4-14. Coolant Loop Concept

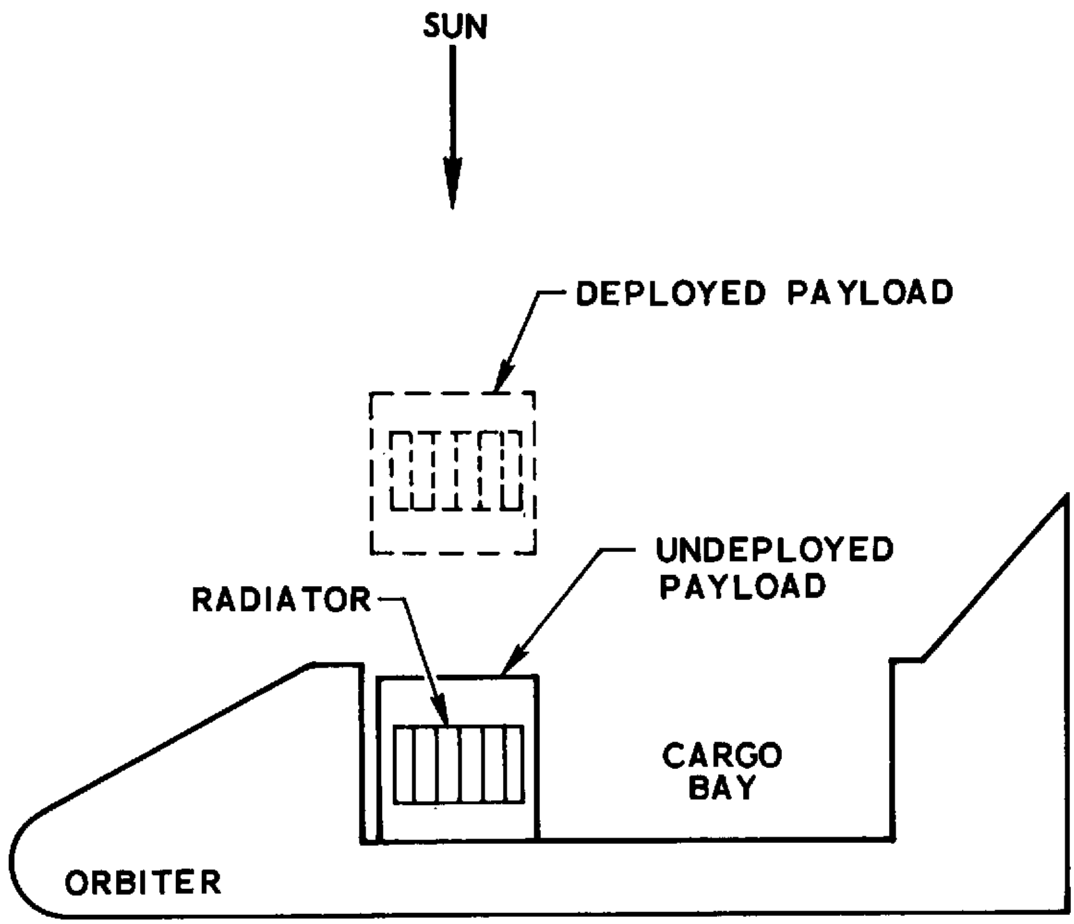


Figure 4-15. Orbiter/Solar Laboratory - Payload Orientation

Thermal control of the structure is accomplished by placing a thermal shell around the truss structure and controlling the temperature of the shell. The truss structure and thermal shell are thermally coupled.

Uniform temperature of the shell is obtained with a heat pipe matrix tied to the inner walls of the thermal shell. The heat pipe consists of 1.3-cm (0.5-in.) standard aluminum tubing and NH_3 as the working fluid. The heat pipe assembly to cover the entire surface area of the ASO is estimated to weigh approximately 47.6 kg (105 lb), of which approximately 3.2 kg (7 lb) is NH_3.

Approximately 2.5 cm (1 in.) of multilayer insulation is placed on the outside of the thermal shell to reduce the solar load, an insulation such as NRC-2 or Superfloc being acceptable. It is assumed that the thermal shell proper is part of the ASO structure.

The thermal weight and power for the ASO is summarized in Table 4-8. The weights depict the thermal inefficiency of the undeployed approach. These austere values were used for the LSO sortie by factoring the weights according to the power increase and telescope aperture.

4.4.5.2 ECLS for ASO Manned Access

To provide the manned access required to repair or adjust the instrument, crew provisions must be included for shirtsleeve IVA operation. A module to enclose the mission equipment and crew for short durations was designed for pressurization whenever the need may exist during the sortie mission. It was assumed that 12.7 m^3 (450 ft^3) of free volume is required, and that sufficient atmosphere for three complete pressurization cycles should be provided for this operation. For only three pressure cycles the simplest and lowest cost approach would be to dump the atmosphere overboard.

Table 4-8. ECLS Weight and Power Summary for ASO (Sortie Mode)

Mode \ Item	Undeployed				Deployed			
	Unmanned		Manned		Unmanned		Manned	
	kg	(lb)	kg	(lb)	kg	(lb)	kg	(lb)
Cooling System	136	(300)	143	(315)	39	(85)	45	(100)
Heat Pipe Assembly	48	(105)	48	(105)	48	(105)	48	(105)
Expendable	36	(80)	36	(80)	16	(35)	16	(35)
Insulation	61	(135)	61	(135)	61	(135)	61	(135)
Thermal Control Weight	281	(620)	288	(635)	163	(360)	170	(375)
Atmosphere (gas)		-	52	(115)		-	52	(115)
(tanks)		-	184	(405)		-	184	(405)
Life Support Plumbing		-	14	(30)		-	14	(30)
Total ECLS Weight	281	(620)	538	(1, 185)	163	(360)	420	(925)
Continuous Power, W	65		65		65		65	
Intermittent Power, W	200		200		200		200	
Maximum Power, W	265		265		265		265	

The atmospheric weight for three cycles is 52.5 kg (115 lb) and the tankage weight is 183.7 kg (405 lb). The ECLS hardware items are assumed to be provided by the orbiter, and the items cited are only the additional expendable weights. No environmental control subsystem is assumed within the LSO because of the relatively short stay time of 2 hrs, plus an open access tunnel.

Life support plumbing referred to in Table 4-8 is the plumbing required to transfer the atmospheric air from the orbiter to the ASO. Additionally, this includes a circulation fan between the orbiter and the ASO. The stay time within the ASO is sufficiently short so that no dedicated humidity control or atmospheric control system is required.

4.4.5.3 Impact of Increasing Orbiter Crew Size to Six Men

The nominal crew size on the orbiter is four men including the pilot, copilot, systems monitor, and payload specialist. Orbiter design provides the capability to accommodate two additional specialists with no system hardware penalty. The consumables associated with the two additional specialists, however, are charged against the payload. The ECLS consumables associated with the two additional specialists are oxygen, food, water, crew provisions, and power. The consumable weights for the two additional specialists for seven days are shown in Table 4-9.

The oxygen requirement is on the order of 0.9 kg (2.0 lb) per man-day. To account for losses and contingencies, however, 1.0 kg (2.2 lb) per man-day was used. It is assumed that the oxygen is an expendable item and not recoverable. CO_2 is removed with a regenerable system and does not result in any consumable weight penalty.

Table 4-9. Incremental Consumable Weight Summary for Two Additional Specialists (Seven Days)

Item	Weight	
	kg	lb
Oxygen	14	31
Food	11	24
Water	53	116
Crew Provisions	18	40
Fuel Cell Reactants	34	76
TOTAL	130	287

Food weight is based on freeze-dried food. The freeze-dried food requirement is about 0.68 kg (1.5 lb) per man-day. If ordinary food is used, food weight will go up by about 1.6 kg (3.5 lb) per man-day and the water weight will go down by the same amount. The weights in Table 4-9 reflect a 10-percent contingency.

Water requirements can be met by either storing expendable water or using fuel cell by-product water. The orbiter crew will probably use stored water; and Table 4-9 shows a 3.4 kg (7.5 lb) per man-day plus 10-percent contingency water weight, based on a freeze-dried diet.

Crew provisions include crew personal effects such as clothing. A weight of 9.1 kg (20 lb) is allocated for each specialist.

It is assumed that each additional person will impose a 250-W requirement on the power system. The fuel cell reactant weight reflecting this is based on a H_2/O_2 fuel cell requiring 0.4 kg (0.9 lb) per kW-hr.

4.4.6 Communication

Payload communications for sortie missions have been examined using the orbiter to ground communication, Intelsat IV, and NASA Tracking and Data Relay Satellite (TDRS). Basically two alternative communication approaches exist in low altitude orbit. The first and the most economical alternative is the direct ground link approach, which utilizes the orbiter communication capability available to the payload. The second alternative involves continuous transmission via satellite.

4.4.6.1 Orbiter Communication (Ground)

The orbiter communications system includes a capability of providing a down link of 256,000 bps digital data, TV, and voice (Ref. 1.2). With this capability one scene requiring 10^7 bits can be transmitted in

40 seconds after signal acquisition and lock-on. If the nominal data pass over a ground station is of 4 to 10 minutes duration, about 6 to 15 scenes can be transmitted. (See Tables 4-10 and Figure 4-16.)

In addition to this amount of scene transmission per orbit the scientist requests that data be transmitted on each orbit. An investigation concerning the number of orbits required to contact the ground stations in a network has been conducted previously in the space escape study (Ref. 4.5). There are several networks that can be used, such as AF/SCF, NASA/MSFN,* and COMSATS. For interface compatibility and security reasons, however, the investigation presented is limited to the NASA network. For this network, the ground stations listed in Table 4-10 were assumed to be available NASA ground stations. Using this network and signal acquisition from 0 deg elevation, the coverage circles for each station are shown in Figure 4-16. With a transponder and prior knowledge of the orbit, communication at this low elevation is assumed to be possible for MSFN. The coverage circles shown are for one- and four-minute communication duration. If the orbiter ground trace just touches the inner circle, four minutes of contact with the station will be obtained. If the trace cuts across the circle, contact will be longer.

The number of orbits required to contact the ground stations for various orbital conditions were computed from the orbital ground trace data. (See Table 4-11.) The results indicate that the MSFN with its large number of ground stations can provide communication contact on each orbit for low and high inclination orbits at 463 km (250 nmi) or higher altitude. It does not appear feasible, however, to have contact on each orbit for those missions in the 463 km (250 nmi) x 55 deg inclination range (space station orbit). It should also be recognized that the sample ground network may not be representative during the Shuttle era. If the number of stations located at certain longitudes and latitudes are eliminated, then the above

*This network is currently designated as STDN (Spaceflight Tracking and Data Network).

Table 4-10. NASA - Manned Space Flight Net (MSFN)

Deep Space Instrumentation Facilities (DSIF)			
Station		Latitude (deg)	Longitude (deg)
MAD	Madrid, Spain	40.45	355.83
CNB	Canberra, Australia	-35.58	148.98
GDS	Goldstone, California	35.35	243.13

Near Space Instrumentation Facilities (NSIF)			
Station		Latitude (deg)	Longitude (deg)
EGL	Eglin AFB, Florida	30.42	273.20
CKY	Cape Kennedy	28.50	279.47
GBI	Grand Bahama Island, Br.	26.62	281.65
ANT	Antigua, Br.	17.02	298.25
ACN	Ascension, Br.	-7.95	345.67
BDA	Bermuda, Br.	32.35	295.35
CYI	Canary Island, Sp.	27.73	344.40
CRO	Carnarvon, Australia	-24.90	113.72
GWM	Guam	13.30	144.73
HAW	Hawaii	22.09	200.33
GYM	Guaymas, Mexico	27.97	249.28
TEX	Texas	27.65	262.62
PRE	Pretoria, So. Africa	-25.95	28.37
CAL	Pt. Arguello, California	34.58	239.42
WHS	White Sands, New Mexico	32.35	253.63
GTK	Grand Turk Island	21.47	288.87
SSI	San Salvador	24.12	285.48

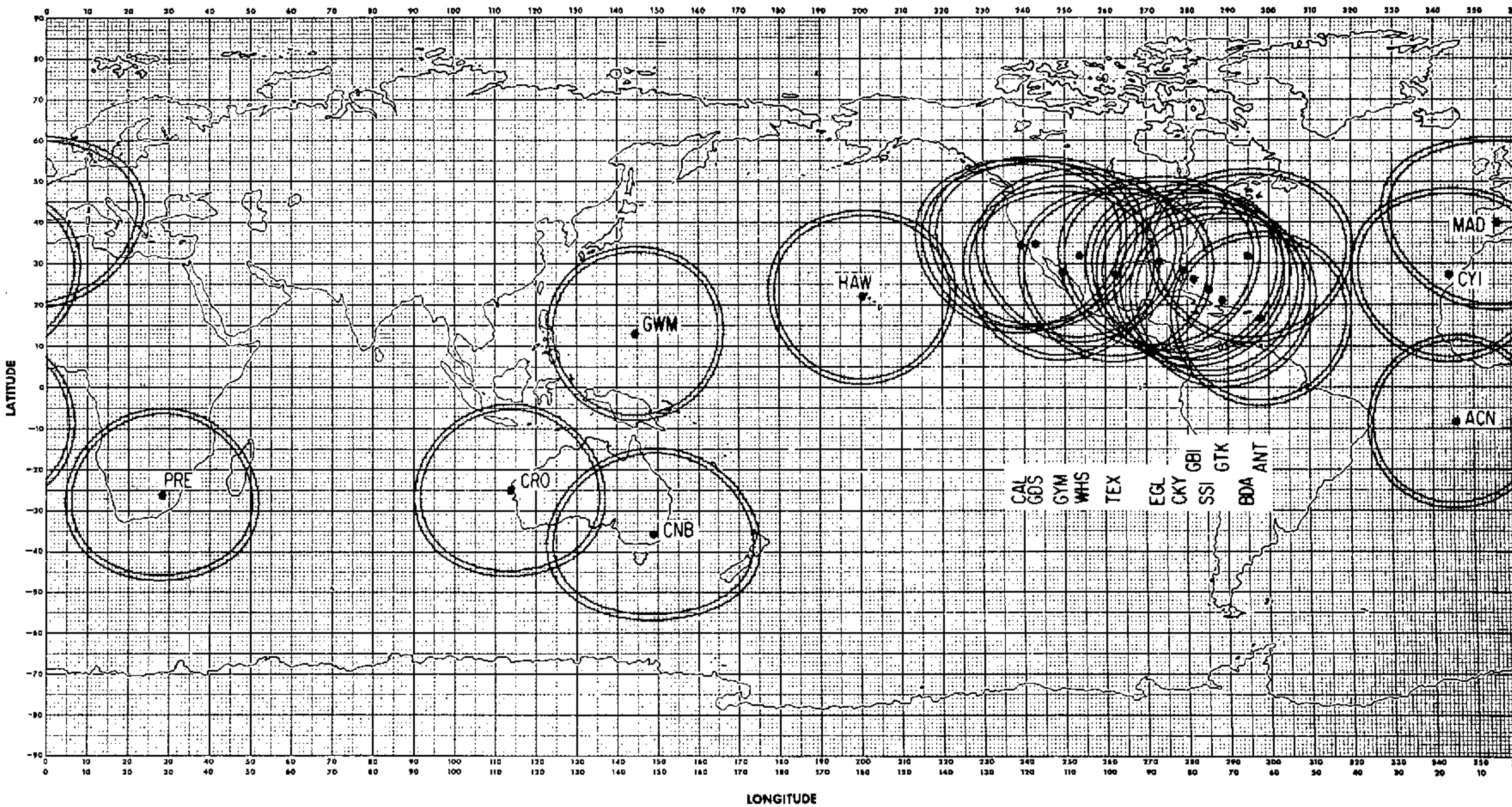


Figure 4-16. MSFN, 463-km (250-nmi) Circular Orbit

Table 4-11. Number of Orbits for Assured Contact Using MSFN

Inclination	Altitude, km (nmi)		
	185 (100)	463 (250)	740 (400)
30 deg	2	1	1
60 deg	3	2	1
90 deg	2	1	1

observation is not applicable. If stations are located in areas of overlapping coverage, then the above observations are applicable. Furthermore, military and Comsat stations may be adapted for ground coverage to augment any decrease in MSFN ground stations.

4.4.6.2 Continuous Communications (Satellite)

RF link power budget calculations for the Intelsat IV and the TDRS are presented in Table 4-12 to accommodate the 10^7 bps data rate (Ref. 4.1). The gain/temperature (G/T) for the Intelsat IV was obtained from Ref. 5.3. G/T and the radio frequency for the TDRS are based on interpretations of Ref. 4.6. The ratio of antenna gain to the absolute temperature of the electronics is a figure of merit of receiving sensitivity. The signal-to-noise ratio (S/N) of 12 dB and the margin of 6 dB were selected on the basis of good engineering practice.

The transmitter power required for communication is influenced by the gain of the antenna used. The power required for communication via the Intelsat IV and the TDRS is called out in Table 4-13 for four sizes of orbiter antennas. The data presented are based on 2 dB losses in the transmitting system, so each antenna/transmitter is 2 dB higher than the effective radiated orbiter power shown in Table 4-12. The antenna beamwidths are also included in the tables. The antenna must be pointed at the satellite with an accuracy of one half the beamwidth in order to maintain the link performance.

Estimates of weight and primary power have been made for several of the transmitters to afford insight into the impact on the orbiter of providing the communication capability. It is estimated that in the Shuttle era a 64-W transmitter at 15 GHz will weigh 9.1 kg (20 lb) and require 320 W of primary power; a 10-W transmitter at 15 GHz will weigh 2.3 kg (5 lb) and require 50 W; and a 4 kW transmitter at 6 GHz will weigh approximately 91 kg (200 lb), not including a liquid cooling system that would be required, and require 12 kW of primary power.

Table 4-12. Power Budget Calculations

	Units	Intelsat IV	TDRS
Effective Radiated Power	dBW	80	59
Polarization Loss (circular to circular)	dB	0	0
Space Loss 42,450 km (23,000 nmi)			
F = 6 GHz	dB	-201	
F = 15 GHz	dB		-209
Received Power	dBW	-121	-150
Receive Sensitivity* (Bandwidth = 2×10^7 cps; S/N = 12 dB)			
G/T = -17 dB/°K	dBW	-127	
G/T = +12 dB/°K	dBW		-156
MARGIN	dB	6	6

*Receive Sensitivity is determined as follows:

$$R_S = \frac{KB}{G/T}\left(\frac{S}{N}\right)$$

or in terms of dB

$$R_S = K + B - G/T + S/N$$

Table 4-13. Antenna/Transmitter Combinations

Antenna Diameter m (ft)	Intelsat IV (6 GHz)				TDRS (15 GHz)			
	Antenna Gain (dB)	Trans Power (dBW)	Trans Power (W)	Beamwidth (deg)	Antenna Gain (dB)	Trans Power (dBW)	Trans Power (W)	Beamwidth (deg)
0.6 (2)	29	53	200,000	6	37	24	252	2.3
1.2 (4)	35	47	50,000	3	43	18	64	1.2
3.05 (10)	43	39	8,000	1.1	51	10	10	0.5
4.6 (15)	46	36	4,000	0.8	54	7	5	0.3

Note: this table is based on 2 dB losses in the transmitting system.

The Intelsat IV receives via a global antenna of 17-deg beamwidth, which is pointed earthward. The 17-deg beamwidth is adequate for earth coverage; however, the angle subtended by a 500-km (270-nmi) orbit from synchronous altitude is about 19 deg. Continuous transmission probably requires the use of three Intelsat IV satellites.

It is apparent from Table 4-13 that there is significantly less impact on the orbiter for communication via the TDRS than for communication via Intelsat IV. The decreased impact on the orbiter of communicating via the TDRS is the result of the improved G/T, which is 29 dB superior to that of the Intelsat IV. The means that will be employed to obtain the TDRS G/T are not known at this time; it is likely, however, that a large part of the improvement over the Intelsat IV will be obtained through the use of a higher gain antenna. If all of the improvements were to be obtained through antenna gain, the TDRS beamwidth would be 0.6 deg and the antenna about 2.4 m (8 ft) in diameter. It is apparent that the TDRS antenna would have to be driven so that it points continually at the orbiter.

The information indicates that the TDRS type satellite will be required for continuous communications. The current Intelsat concept is designed for large and powerful ground stations, which is not consistent with orbiter transmission capability. Furthermore, the current Intelsat IV allocates frequencies to each user, which implies that the orbiter must schedule downlink frequencies to earth stations being overflown. The TDRS will have similar operational problems, such as scheduling the frequencies and pointing with other user programs. Both methods must switch either from earth station to earth station or from TDRS satellite to TDRS satellite. Thus, the area of continuous communication is not clear at this time. Further analysis will be required as data on relay satellites become better defined for the Shuttle era.

4.4.7 Weight Estimates

Weight estimates were based on available data sources and subsystem analyses performed in this study. Basically, the conceptual design provided the envelope dimensions and the hardwares envolved. Estimates of the structural weight used, the structural unit weights obtained from Ref. 4.4, and the structural size and components were obtained from the conceptual layout. The reference unit weights were examined and found to be representative of other study results using similar structures.

The resultant structural equations are:

Unpressurized structure weight = 0.98 g/cm^2 (2.0 lb/ft^2) x (wetted area)
Pressurized structure weight = 1.51 g/cm^2 (3.1 lb/ft^2) x (wetted area)
Equipment supports = 0.247 x (equipment weight)
Meteoroid protection = 0.88 g/cm^2 (1.8 lb/ft^2) x (wetted area)

The wetted area was determined from the conceptual layouts.

The stabilization and control weights were based on data in section 4.4.4 and Table 4-7 concerning CMGs, torquers, and electronics. The gimbal structure weights, which amounted to approximately 50 percent of the gimballed weight, were derived from Ref. 4.4. For the tethered and free flyer configuration, the control weights were based on data supplied in Volume II of this report. The propellant quantities for configurations C and D were oversized for the seven-day missions.

The environmental control weights were developed in section 4.4.5 and summarized in Table 4-8 for the ASO mission equipment case. For the LSO case these weights were factored in accordance with electrical power, solar energy, and size. The cooling weight for the ASO concept assumed the use of the orbiter radiator. This assumption reduced the cooling weight from 136 kg (300 lb) to 15.9 kg (35 lb), which is the estimate for hook-up only.

The telemetry, tracking, and command weights for the free flyer, configuration D, are based upon the data supplied in Volume II of this report. The weights shown for the other designs represent estimates for recording devices and feedback instrumentation only.

The electrical weights are for power distribution leads except that configuration D, a free flyer, carries its own oriented solar array which provides 1 kW of power. In all other cases it is assumed that the orbiter can supply the electrical load. Normally the orbiter produces 3 kW average with a 6 kW peak electrical power, which should be sufficient for the intermittent peak loads for the LSO.

The mission equipment weights were obtained from Tables 4-2 and 4-3. The overall weight summary is shown in Tables 4-14 and 4-15 for the LSO and ASO configurations. A contingency allowance of 10 percent of the spacecraft weight is included to account for normal weight growth.

The pallet, displays, CMGs, and payload support are those items that are in addition to the equipment supplied by the orbiter, such as manipulators, and standard control and checkout equipment. The pallet weight accounts for only that portion required to support the module during ascent and descent phases. The display is for the unique monitoring equipments. The CMG unit is to augment the orbit stabilization and control subsystem. The payload support item is for the tilt table used in deploying the payload.

4.5 OPERATIONS

This section presents the comments supplied by the scientist on the solar observation operations, operations comments on the various designs postulated in this study, and a typical crew timeline. Analysis of the overall payload and Shuttle operations is presented in Section 2 of this volume.

Table 4-14. Large Solar Observatory - LSO Mission Equipment Weights

Subsystem	(A) Hardmate		(B) Gimballed		(C) Tethered		(D) Free Flyer	
	kg	(lb)	kg	(lb)	kg	(lb)	kg	(lb)
Structure	2,651	(5,845)	6,213	(13,697)	2,728	(6,015)	2,882	(6,353)
Basic	1,438	(3,170)	1,438	(3,170)	1,515	(3,340)	1,668	(3,678)
Equipment Supports	1,213	(2,675)	1,213	(2,675)	1,213	(2,675)	1,213	(2,675)
Gimbal Structure		--	3,562	(7,852)		--		--
Environmental Control	564	(1,244)	564	(1,244)	584	(1,288)	624	(1,376)
Cooling	160	(352)	160	(352)	160	(352)	160	(352)
Heat Pipe	160	(353)	160	(353)	168	(371)	185	(408)
Expendables	39	(86)	39	(86)	39	(87)	41	(90)
Insulation	206	(453)	206	(453)	217	(478)	239	(526)
Guidance and Navigation		--		--	816	(1,800)	816	(1,800)
Gimbal Electronics		--	728	(1,605)		--		--
Attitude Control		--		--	113	(250)	227	(500)
Propellant		--		--	79	(175)	159	(350)
Inerts		--		--	34	(75)	68	(150)
Telem and Communications	23	(50)	45	(100)	68	(150)	172	(380)
Electrical Power	154	(340)	154	(340)	154	(340)	730	(1,610)
Distribution	154	(340)	154	(340)	154	(340)	186	(410)
Solar Arrays		--		--		--	544	(1,200)
Contingency (10%)	339	(748)	766	(1,689)	446	(984)	545	(1,202)
Mission Equipment	4,910	(10,825)	4,910	(10,825)	4,910	(10,825)	4,910	(10,825)
Payload Weight	8,642	(19,052)	13,381	(29,500)	9,821	(21,652)	10,907	(24,046)
Pallet	1,212	(2,672)	1,212	(2,672)	1,284	(2,830)	1,416	(3,122)
Displays	318	(700)	318	(700)	318	(700)	318	(700)
Control Moment Gyros	1,588	(3,500)		--		--		--
Payload Support	113	(250)	113	(250)				
Total Weight	11,872	(26,174)	15,024	(33,122)	11,422	(25,182)	12,641	(27,868)

Table 4-15. Austere Solar Observatory - ASO Mission Equipment Weights

Subsystem	(E) Unmanned Non-Deployed		(F) Manned Deployed		(G) Manned In GPL		(H) Manned Add On to GPL	
	kg	(lb)	kg	(lb)	kg	(lb)	kg	(lb)
Structure	1,864	(4,110)	3,431	(7,565)	2,041	(4,500)	3,826	(8,435)
Canister	274	(605)	665	(1,465)	261	(575)	261	(575)
Equipment Supports	399	(880)	399	(880)	399	(880)	399	(880)
Tunnel			213	(470)				
Door Modification			91	(200)	1,381	(500)	227	(500)
Pressure Shell							1,091	(2,405)
Gimbal Structure	1,191	(2,625)	1,762	(3,885)	1,154	(2,545)	1,154	(2,545)
Meteorite Protection			302	(665)			694	(1,530)
Environmental Control	161	(355)	420	(925)	984	(2,170)	1,100	(2,425)
Cooling	16	(35)	45	(100)	45	(100)	45	(100)
Heat Pipe	48	(105)	48	(105)	48	(105)	48	(105)
Expendables	36	(80)	16	(35)	16	(35)	16	(35)
Insulation	61	(135)	61	(135)				(in structure)
Atmos. Expended			52	(115)	191	(420)	216	(475)
Atmos. Plumbing			14	(30)	14	(30)	14	(30)
Atmos. Tanks			184	(405)	671	(1,480)	762	(1,680)
Guidance and Navigation								
Gimbal Electronics	93	(205)	93	(205)	93	(205)	93	(205)
Attitude Control								
Telem and Communications	27	(60)	27	(60)	27	(60)	27	(60)
Electrical Distribution	181	(400)	91	(200)	91	(200)	91	(200)
Contingency (10%)	228	(503)	406	(895)	324	(715)	513	(1,130)
Mission Equipment (Austere)	1,619	(3,570)	1,619	(3,570)	1,619	(3,570)	1,619	(3,570)
PAYLOAD WEIGHT	4,174	(9,203)	6,087	(13,420)	5,180	(11,420)	7,269	(16,025)
Remote Controls	91	(200)						
Pallet	356	(785)	356	(785)			463	(1,020)
Displays	227	(500)	227	(500)	227	(500)	227	(500)
Payload Support			113	(250)	113	(250)	113	(250)
Total Weight	4,848	(10,688)	6,783	(14,955)	5,520	(12,170)	8,072	(17,795)

4.5.1 Comments by Solar Scientific Community

The solar scientific community, in general, has no desire to follow the normal engineering practices of rigid schedules and early definitions of mission objectives and mission equipment. Some specific comments are:

(1) Mission equipment management should be separated from payload program management. (Scientists would like to release their equipment when they are ready.)

(2) Launch schedules are generally not critical. (Scientists are willing to wait several weeks.)

(3) Integration of mission equipment should be flexible. (Sensor development progress is uncertain; scientists would like to be able to change equipment as better units or a higher priority experiment develops.)

(4) Access to mission sensor near launch time is desirable. (Current large space experiment program access is three months from launch.)

(5) Mission specialists should be limited to two technicians. (Scientists will be on the ground as observers to provide instructions via a communications link for five to ten minutes on each orbit.)

(6) On-orbit calibration is desirable. (Certain sensors such as X-ray and UV require adjustments.)

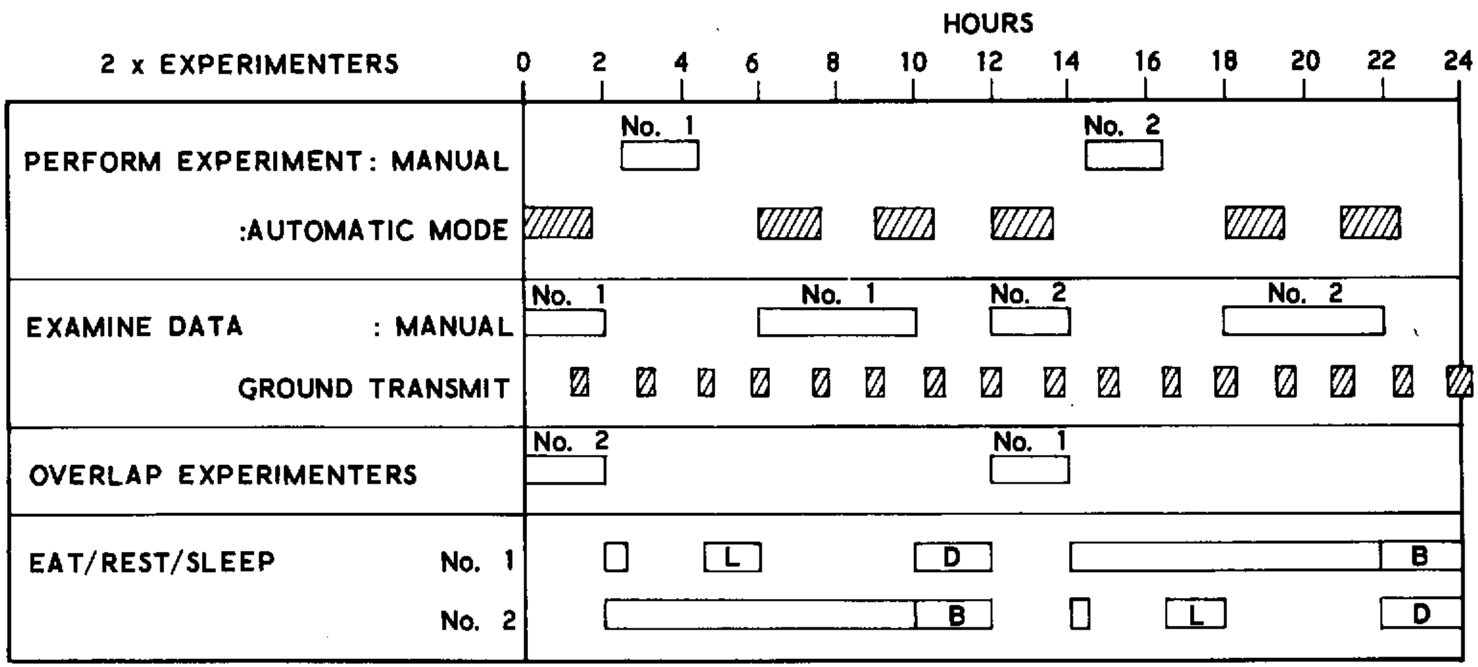
(7) Manned access to mission equipment is desired to repair, readjust, or recalibrate instruments during orbiter operations.

These comments generally can be accommodated by the sortie, providing certain design and operational requirements are implemented. The sortie payload module or pallet should be designed like a general-purpose sortie with access on orbit by technicians. The mission equipments should be selected and installed as they become available for launch. It appears that it is more desirable for the mission equipment to wait for a launch than for the sortie to wait for the mission equipment. The general-purpose sortie should be capable of interfacing with astronomy, space physics, earth observation, or communication type disciplines with minor modifications. The current maximum crew size of two crewmen and two specialists appears consistent with scientists' needs for two technicians. It appears, however, that payload access close to launch should be provided in the Shuttle countdown sequence.

4.5.2 Crew Timeline

The orbiter operational procedure according to the Program Requirements Document, Level 1 (Ref. 1.1) states that the "orbiter reaction control system shall be capable of pointing an attached, exposed payload continuously for one orbit every other orbit for one 24-hour period per mission at any ground, celestial, or orbital object within $\pm$ 0.5 degree." This restriction could imply a total of eight complete orbits, or a total of 12 hours of solar sightings, per mission. This would limit the sorties to a total of 12 hours of experimentation over seven days. The free flyer approach described in this section would not have such a restriction since the orbiter does not need to hold its pointing.

If this restriction on attached payloads can be modified "to only every other orbit and expendables to continue this cycle to greater than 24 hours shall be provided by the payload," the experimentation time will be substantially increased. On the basis of such a case, a crew timeline was developed to examine the type of timelines for four and six total crew sizes. (See Figures 4-17 and 4-18.) A typical schedule that would be required to monitor the large solar observatory on every other cycle sighting capability over five days is illustrated.

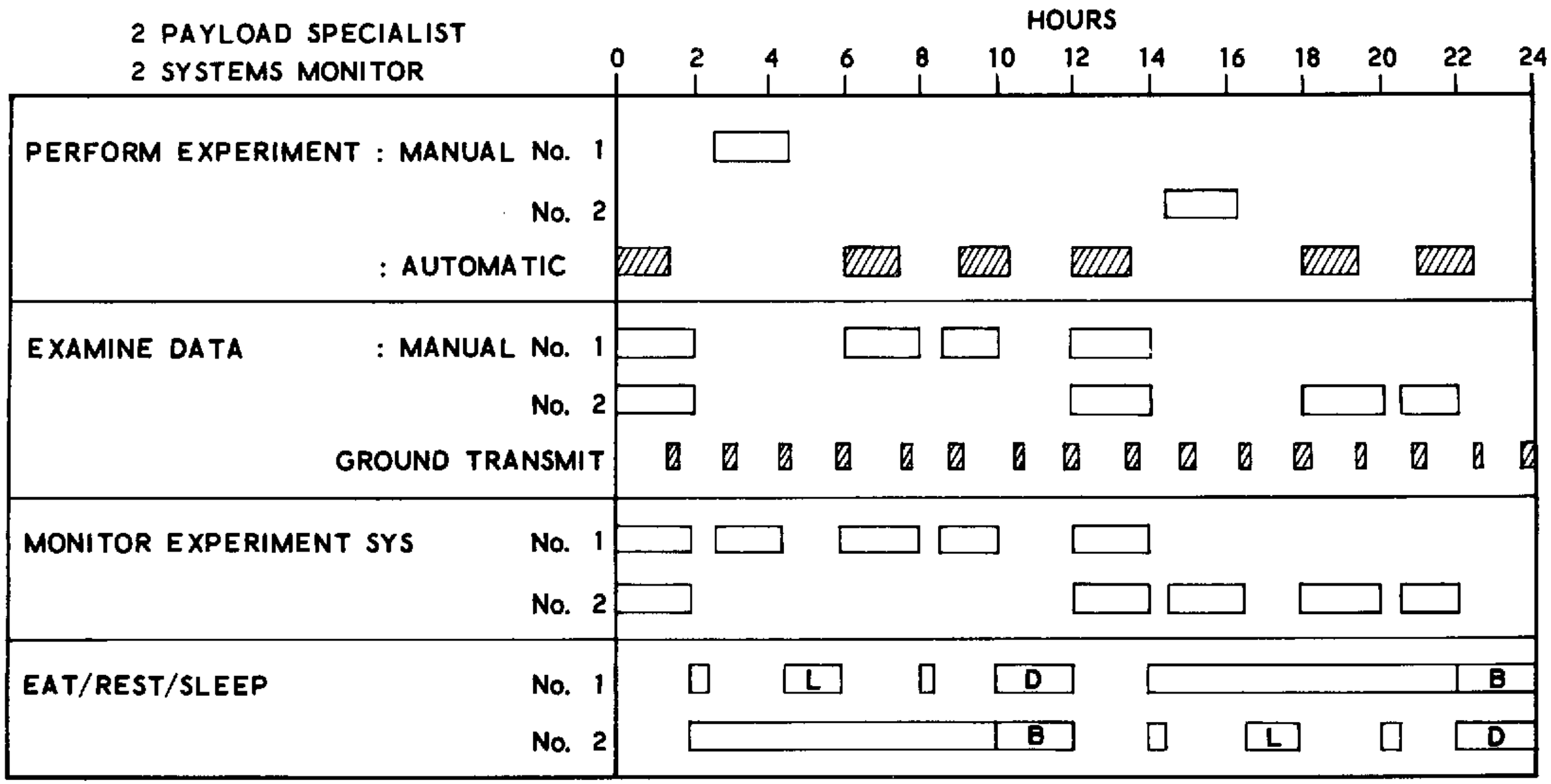


CONDITION:

CREW: 1 PILOT } PERFORM HOUSEKEEPING FUNCTIONS
1 CO-PILOT }
2 EXPERIMENTERS PERFORM SCIENTIFIC TESTS & SYSTEM MONITOR
4 TOTAL

7 DAY MISSION; 1 DAY SET-UP, 5-1/2 DAY EXPERIMENT, & 1/2 DAY SET-DOWN

Figure 4-17. Typical Sortie On-Orbit Crew Timeline



CONDITION: CREW:

1 PILOT	}	HOUSEKEEPING FUNCTIONS
1 CO-PILOT		
2 SYSTEM MONITOR	}	PAYLOAD FUNCTION
2 PAYLOAD SPECIALIST		
6 TOTAL		

MISSION DURATION:

1	DAY FOR TEST SET-UP AND OUTGASSING
5-1/2	DAY EXPERIMENT
1/2	DAY FOR RETURN
7	DAYS

Figure 4-18. Typical Sortie On-Orbit Timeline

In both cases the pilot and co-pilot were assumed to provide the housekeeping functions during orbital operations. The mission specialists would consist of two experimenters in the four crew size case. They would operate on basically a 14-hour cycle, which includes two hours for crew overlap time. Most of their on-duty time would be directed to data examination for ground transmittal. This approach assumes that the solar sensors can generally operate in the automatic mode. The manual mode is for calibration, redirect sightings, and observations.

The six-man crew setup consists of a pilot and a co-pilot for housekeeping functions, two system monitors, and two payload specialists. The system monitors are system engineers/technicians and the payload specialists are scientists/laboratory assistant types. One man from each specialty would form the team, which would operate on a 14-hour schedule for five days. The schedule would be tight and extensive crew training would be required if this cycle were to be followed. Crew training is costly and time-consuming on the part of the scientist; hence crew/machine interface should be examined in more detail. The feasibility of such schedules should be examined by the scientists and payload engineers. The schedule also suggests that there is little time for the experiment to be manned for calibration, maintenance, and adjustment. The manned access would, however, be beneficial in the case where there is only 12 hours of experimentation time available. Manned access would maximize the effectiveness of the 12 hours available for experimentation.

4.5.3 Operations

The LSO and ASO configurations were briefly examined from an overall operations aspect. The orbiter/payload interface, ground test, on-orbit checkout, and orbital operations areas were grossly evaluated to rank the concepts operationally. The critical feature in the orbiter/payload interface area is the expected mechanical and electrical tie-in between the orbiter and the payload. The ground tests encompass the payload tests, installation and calibration, and subsystem test. The on-orbit checkout includes deployment and checkout. Orbital operation is the actual mission operations phase. The backup operations include the capability to correct unplanned situations that can occur during an operation. In each category, elements were ranked as simple (+), intermediate (0), and complex (-1). These rankings are summarized in Table 4-16 for all of the configurations.

For the LSO sortie cases, the hardmate (configuration A) is ranked as the simplest operational approach. The mechanical interface and umbilicals connections should provide direct attachment. The payload package should be deployable in the ground simulator tests, thus permitting complete system test on the ground. The on-orbit checkout should duplicate the ground checkout, and the operations phase should also duplicate ground tests. The tether (configuration C) does not provide for standard test procedures. The orbiter to payload umbilical will be played alongside the manipulator in the stored and deployed arrangement. The manipulator will not be able to deploy the ASO in the ground simulated tests to completely duplicate operational tests on the ground. The gimbal (configuration B) and free flyer (configuration D) are intermediate cases in the overall operations ranking.

For ASO sortie cases, the unmanned gimbal (configuration E) approach appears to have the highest ranking because of its simplicity, being unmanned and non-deployable. The capability for having manned access increases the complexity in the interface hook-up and ground test procedures, but it improves the backup operation by the shirtsleeve IVA capability. The unmanned versions require a manipulator or EVA to address anomalies or failures. It is expected that sharing the GPL will involve the scheduling

Table 4-16. Operational Ranking

Configuration	Orbiter/ Payload Interface	Ground Test	On-Orbit Checkout	Operation	Backup Operation	Total
LSO						
A hardmate unmanned ops deployed	+	+	+	+	-	3
B gimbal unmanned ops deployed	0	0	+	+	-	1
C tethered unmanned ops deployed	-	-	0	+	-	-2
D free flyer unmanned ops deployed	+	0	0	+	-	1
ASO						
E gimbal unmanned ops undeployed	+	+	+	+	-	3
F gimbal manned ops deployed	-	0	0	+	+	1
G gimbal integrated GPL	0	0	0	-	+	0
H gimbal attached GPL deployed	0	+	0	-	+	1

Note: (+) = +1, (0) = 0, (-) = -1

of on-orbit maneuvers with other ongoing scientific experiments. This aspect can also be involved during the ground test when the ASO is an integral part of the GPL. When the ASO module is separate from the GPL the integration and testing procedures are reduced because many ground operations can be performed independently. In totaling these rankings the gimbal (configurations F and H) with separate manned modules appears to have better operational characteristics than the completely integrated case (configuration G) due to the relative independence of the GPL.

These rankings, totaled in Table 4-16, should be considered at this time as operational indicators. Furthermore, these estimated operational rankings were based on conceptual layouts and were made without detailed operational analysis.

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4.1 Scientific Objectives and Instrument Performance Criteria for a Large Solar Observatory, E. B. Mayfield, et al, ATR-72(7268)-1, The Aerospace Corporation (August 1972).

4.2 1.5-Meter Photoheliograph Concept for Large Solar Observatory, 72-8212-3, ITEK/Optical Systems Division (3 April 1972).

4.3 Control Moment Gyro Selection and Design Criteria, AIAA Paper No. 70-976, G. Ouclair and R. Wells, General Electric Company (August 1970).

4.4 Mass Properties Document, Research and Applications Module (RAM) Phase B Study, GDCA-DDA71-006A, General Dynamics Convair (April 1972).

4.5 Survey of Communication and Navigation Concepts for Space Escape System, TOR-0200(4525-04)-03, T. Shiokari, The Aerospace Corporation (June 1969).

4.6 Statement of Work, Tracking and Data Relay Satellite System, Goddard Space Flight Center, Greenbelt, Maryland (11 May 1971).

5. Automated
Spacecraft

5. AUTOMATED SPACECRAFT

The Intelsat IV satellite was selected for the automated spacecraft mission because it represents an applications type payload requiring an energy stage. Also, it was expected that design information and costs on this typical communications satellite would be documented and complete enough to permit these studies to be pursued.

In the NASA 1972 mission model for the period 1979 to 1990, 110 out of a total 287 payloads (40 percent) are to be launched into orbits beyond low earth orbits. Specifically, they consist of satellites requiring upper stages to function in orbits such as synchronous equatorial, planetary, high elliptical, or high altitude [<1, 100 km (600 nmi)] orbit. Forty-five payload launches of the 110 (16 percent) are communication satellites. The communication mission represents 16 percent of NASA satellite launches and 40 percent of NASA launches requiring an upper stage in the 1972 NASA mission model.

5.1 SATELLITE DESCRIPTION

A description of Intelsat IV, including the general location of each unit, is presented in Figure 5-1. The overall height is 5.3 m (17.5 ft) and the diameter of the solar drum is 2.7 m (7.75 ft). The satellite is a spin-despun type satellite with the antennas and communication subsystem located on the despun platform. The mission equipment is all electronic. A more detailed description of the satellite is given in section 5.3.1.

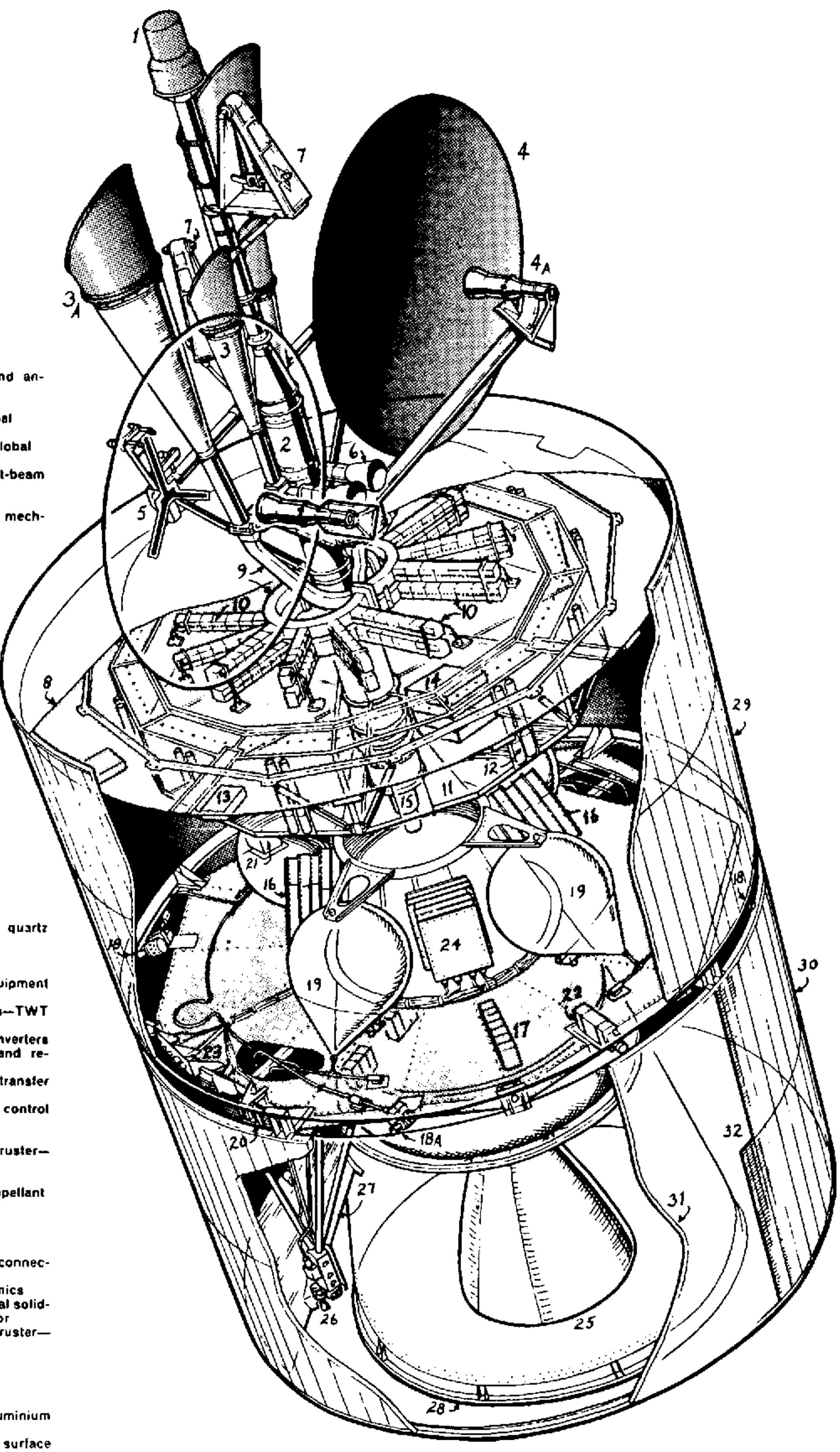
The satellite is sufficiently representative in design characteristics of system demonstration satellites to illustrate, on a relative payload program cost basis, the tradeoff for:

(1) Variation in satellite design life

(2) Shorter length satellite (two satellites per launch vs one)

(3) Spacecraft adapted to mission equipment design changes vs new satellites

1 Telemetry and command antenna
2 Antenna mast structure
3 Receive antenna, global coverage
3a Transmit antenna, global coverage
4 Transmit antenna, spot-beam dish
4a Feeder horn
5 Antenna positioning mechanism
6 Telemetry horn antenna



7 Nutation dampers (2)
8 Forward sun shield, quartz mirror material
9 Waveguide
10 Output multiplexers
11 De-spun electronic equipment platform
12 Travelling wave tubes—TWT (12 x 2)
13 TWT power supply converters
14 Telemetry and command repeater electronics
15 Bearing and power transfer assembly (Bapta)
16 Electrical power control electronics
17 Battery pack (2)
18 Attitude control thruster—radial (2)
18a Spin-up jet (2)
19 Hydrazine monopropellant tanks (4)
20 Sun sensor (3)
21 Accelerometer
22 Earth sensor (3)
23 Spun wiring harness connections
24 De-spin control electronics
25 Nozzle—Aerojet General solid-propellant apogee motor
26 Attitude control thruster—axial (2)
27 Aft thermal barrier
28 Booster adapter
29 Forward solar panel
30 Aft solar panel
31 Resin-bonded aluminium honeycomb drum
32 Solar cells bonded to surface of drum.

Figure 5-1. Intelsat IV Description

A comparison of the design characteristics of Intelsat IV, Tracking and Data Relay Satellite, and System Test Satellite is shown in Table 5-1.

5.2 SYSTEM PLANS AND TRAFFIC MODELS

The programs selected from the NASA 1972 traffic model as being representative were the Tracking and Data Relay Satellite (TDRS) and System Test Satellite. TDRS represents an on-going system demonstration while the System Test Satellite programs represent system demonstrations limited in duration by the satellite lifetimes. It is of interest, therefore, to determine the tradeoffs between payload design approaches and program costs. The influence of communications advancement and increased communications capacity (user demand) are analyzed separately.

The TDRS is a communication system designed to develop and demonstrate a world-wide tracking and data acquisition satellite to support low earth orbiting space programs including the Space Shuttle. The baseline traffic as it appears in the June 1972 NASA mission model is shown in Table 5-2, Case 1, with an initial launch by an expendable booster in 1978. In the five years between 1978 and 1983 three new communication satellites are to be put into operation to accommodate communication demands for new technology, broader bandwidth, and improved systems. This is to be repeated in 1989.

Case 2 represents the situation where the plans are revised to maintain the 1978 vintage satellite in operation for twelve years. Case 3 repeats the baseline mission equipment changeout schedule but uses the same spacecraft design for eleven years. The spacecraft is adapted to block changes in mission equipment. This concept will adapt to gains in mission equipment technology, demands in communications, and the needs of users.

Table 5-1. System Demonstration Traffic for Communication Program Trade-offs

Cases	Payloads	78	79	80	81	82	83	84	85	86	87	88	89	90
	Tracking and Data Relay Satellite													
1	Baseline	3					3						3	
2	Same Satellites	3												
3	New mission equipment (only)	3					3						3	
4	Same satellites													
	(increasing demand) (2 in	3												
	10 yrs)						1							
													2	
5	Same satellites													
	(increasing demand) (3 in	3												
	10 yrs)						3							
													3	
	System Test Satellites													
1	Mission #1			1	1									
	Mission #2					1	1							
	Mission #3							1	1					
	Mission #4										1	1		
2	Retrieve Satellite								1	1				

Table 5-2. Comparison of Satellite Design Characteristics[1]

Design Characteristics	Intelsat IV	TDRS	System Test Satellite
Launch weight kg (lb)	640-668 (1,420-1,473)	1,043-798 (2,300-1,760)[2]	907-1,297 2,000-2,860
Overall length m (ft)	6.7-5.3 (22-17.5)	5.2 (17)	4.6 (15)
Diameter m (ft)	2.7 (9)	3.05 (10)	3.7 (12)
Mission Equipment Type	SHF	HF, UHF, LASER	VHF, SHF, EAF
Mission Equipment Weight kg (lb)	159 (350)	272-59 (600-219)	272-143 (600-315)
Stabilization & control type	Dual Spin	Dual Spin	Spin or 3 Axis - 3 Axis

Notes: (1) Characteristics in ranges reflect data from 1971 and 1972 NASA Payload Data Book (Ref. 5.5 and Vol. II of this report.)

(2) Weight includes apogee motor.

Cases 4 and 5 address only the increases in communication demands by placing more of the same design satellites into orbit. Neither technology advancements nor changes in user systems are represented in this traffic model.

The system test satellites are communications programs developed to demonstrate satellites for such potential users as law enforcement, post office, air traffic control, maritime service, and land traffic control organizations. For each such user two satellites are planned for launch in successive years. Upon successful demonstration, the satellites are to be given to the particular user agency. There are no plans to repeat these programs; each start will feature different mission objectives. (This traffic model is detailed in Table 5-2.)

The tradeoff with the Shuttle/Tug is to investigate the retrieval and reuse of the spacecraft. It is possible to retrieve Mission #1 in 1985 and 1986 since these satellites will have reached the five-year mean mission duration (MMD). The spacecraft portion can then be refurbished and reused for Mission #4. Another possibility is the continued use of these satellites after refurbishment by the non-NASA user. The former use was assumed for this analysis.

In addition to the payload traffic, additional unscheduled payload flights for random satellite failures are included. This was determined from a reliability analysis of the satellite design. It is shown graphically in Figure 5-2. The design and failure rates of these satellites are discussed in the following section. The expected number of launches is a function of the number of years for one satellite on orbit. This is shown for two Intelsat IV designs, the baseline and weight optimized configurations. The baseline represents the baseline expendable and the minimum modification configurations; the weight optimized includes the redesigned configuration.

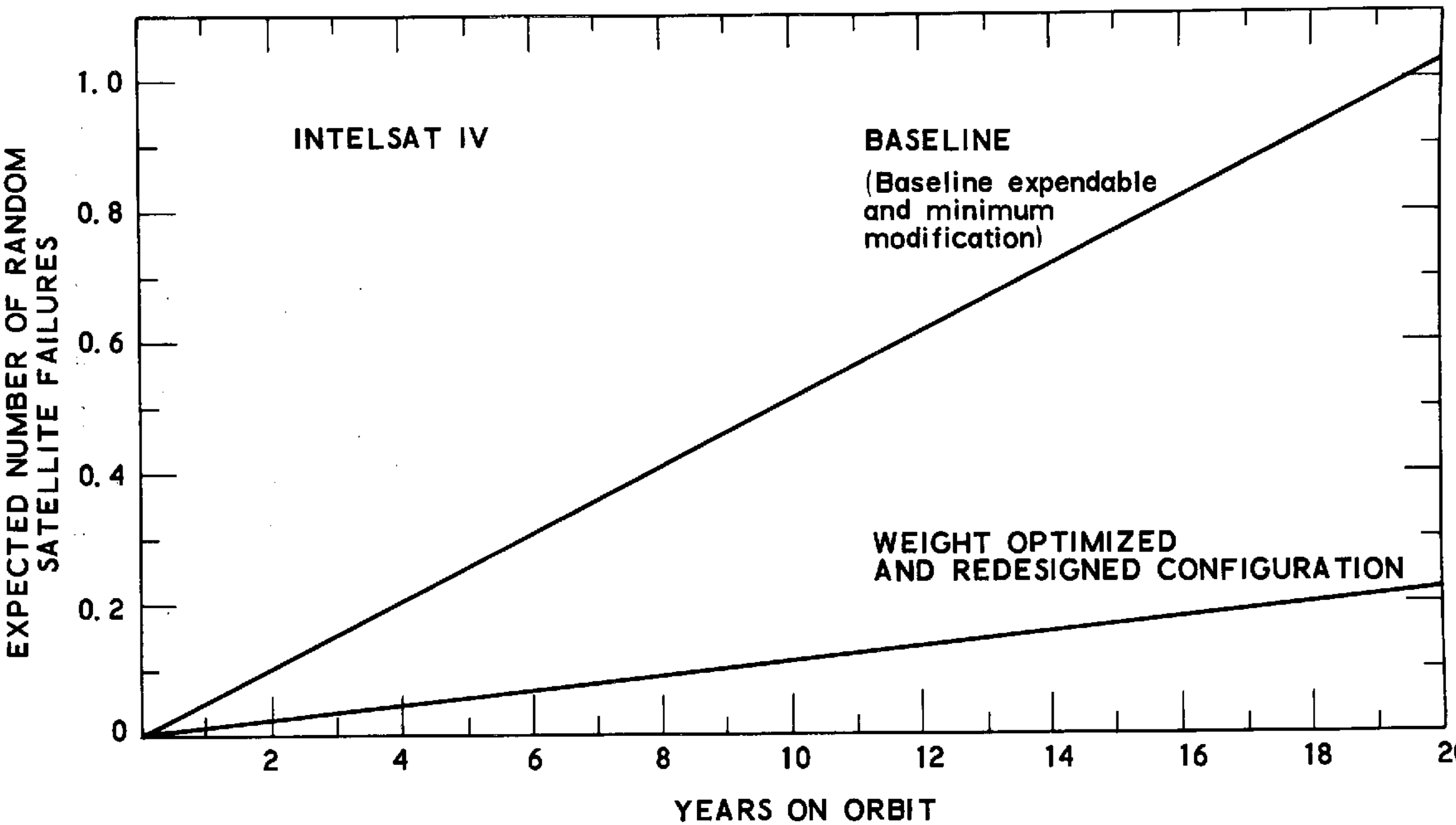


Figure 5-2. Expected Number of Intelsat IV Launches for Random Failures

5.3 DESIGN

A design study was performed to investigate the feasibility of adapting the Intelsat IV satellite for Shuttle operations in which a Tug would be used for deploying and retrieving a payload between the Shuttle and synchronous orbit. The major design problems are to determine the feasibility of rendezvous, remotely retrieving a spinning satellite in synchronous orbit, and achieving rendezvous with the orbiter in low earth orbit. This design effort was limited to the modifications required to adapt an existing satellite, referred to as baseline expendable, to Shuttle/ Tug operations and to reconfigure the satellite to better utilize the orbiter characteristics. The designs are sufficient only for costing and for demonstrating feasibility. Aspects of extending the life of the satellite design required detail analysis because of their influences on cost. The extension of design life was achieved in this study by standby redundancy of components and was determined analytically in the form of redundancy level.

5.3.1 Baseline Design

The Intelsat IV is launched by Atlas/Centaur into a 370 x 40,700-km (100 x 22,000-nmi) transfer orbit. At apogee of an equatorial crossing the Intelsat IV solid rocket motors (SRMs) are fired to achieve final synchronous equatorial orbit. The Intelsat is separated from the Centaur and the satellite is spun-despun within two sec of separation. After the satellite's injection into synchronous orbit the spin axis is oriented to the plane of the orbit. The spin speed of the satellite on station is approximately 60 rpm.

A block diagram of the satellite communications subsystem is shown in Figure 5-3. The RF signal at approximately 6 GHz is received by a global receiving antenna. The signal is amplified and translated to the

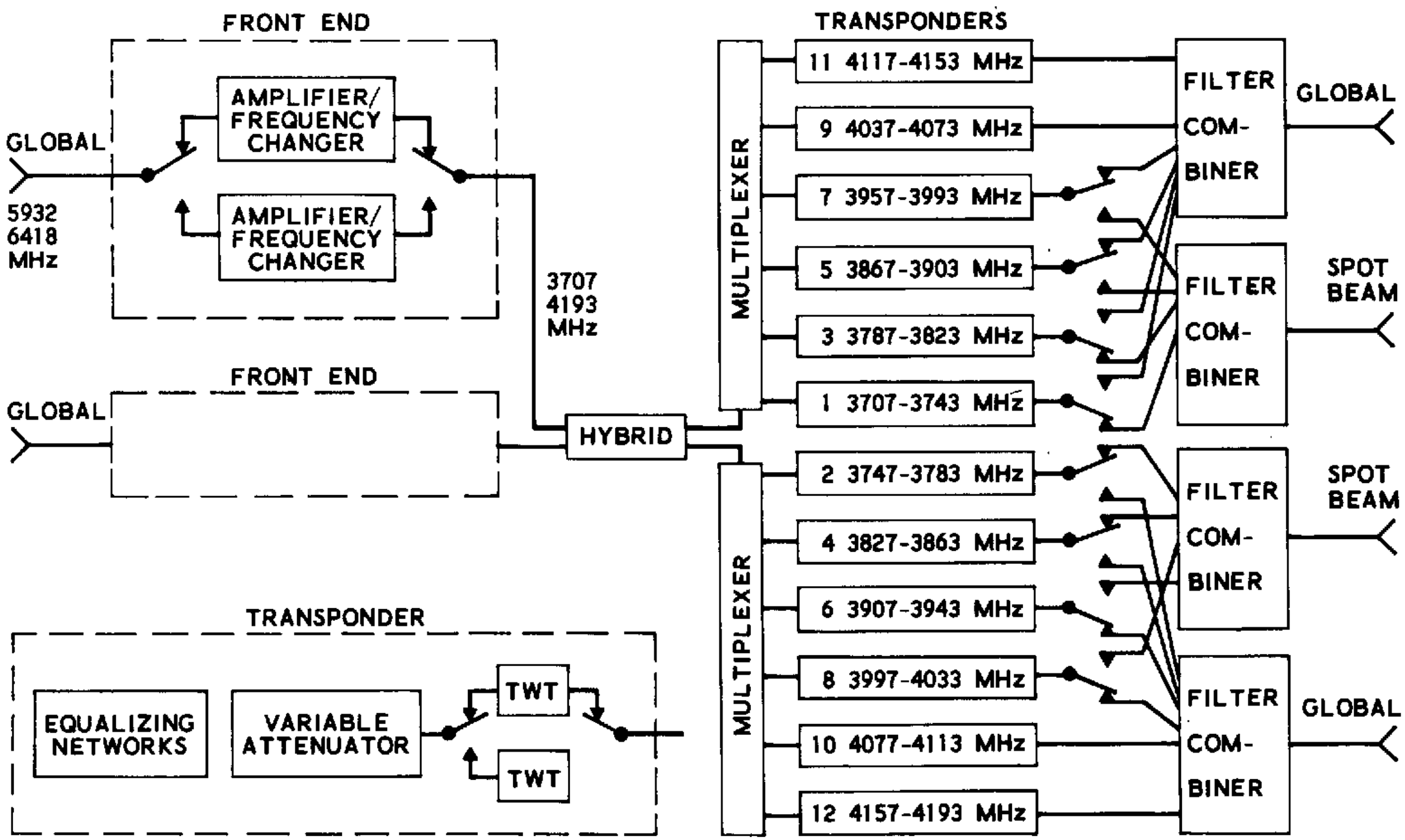


Figure 5-3. Intelsat IV Communication Subsystem

downlink frequency of approximately 4 GHz in the amplifier/frequency changer. The front end and the amplifier/frequency changer in use are selected by ground command. The signal is distributed for future amplification to the 12 transponders, each of which has a bandwidth of 36 MHz. Each transponder has redundant TWTs, controlled by ground command. Also, each transponder contains a variable attenuator similarly controlled by ground command so that the correct TWT operating point can be selected for various conditions of spot beam coverage, global coverage, or multiple/single carrier operation.

The outputs from the 12 transponders are connected to the transmitting antennas through the filter/combiners in such a manner that transponders 9 through 12 are permanently connected to global coverage antennas, while transponders 1 through 8 can be switched by ground command either to a global coverage antenna or to a spot beam antenna. Transponder assignments are made so that the lower frequencies are assigned to the spot beam antennas and the higher frequencies are assigned to the global antennas.

The earth coverage antennas have a beamwidth of 17 deg, which permits coverage of about one third of the earth's surface. The spot beam antennas are approximately 127 cm (50 in.) in diameter and have a beam-width of 4.5 deg. This corresponds to a circle with a diameter of about 2,775 km (1500 nmi) directly below the satellite. As the spot beam antennas are pointed away from the center of the earth, the area of coverage increases and the shape departs from a circle.

The spinning portion of the satellite consists of the cylindrical solar array, solid propellant apogee motor, attitude control subsystem, and structural adapter. The slip ring and torquer for the despun platform are located within the bearing and power transfer assembly. The seven antennas, multiplexes, traveling wave tubes, and repeaters are located on the despun platform. The seven antennas consist of two global receivers, two global transmitters, two spot beam transmitters, and one omni antenna. The two spot beams are 127 cm (50 in.) in diameter and are individually steerable over the visible portion of the earth.

The detail parts list that was used in this study is shown in Table 5-3, which also shows the failure rate and weight of each unit used in the reliability analysis. The general locations of these units are revealed in Figure 5-1. The diameter of the solar drum is 19.7 cm (7.75 in.) and the overall height is 5.3 m (17.5 ft). The length of the cylindrical solar panels encircling the satellite is 2.8 m (9.28 ft). The liftoff weight including the apogee kick motor is 1,407 kg (3,103 lb).

In addition to making studies of the drawings and summary reports (Ref. 5.3), Aerospace staff members toured the Intelsat IV assembly and test area, and participated in discussions on the satellite's testing procedure. The current Intelsat provides for component accessibility through removal of the large spot beam antennas and the forward solar drum from the despun assembly. The solar drums are mated at the mid-section of the cylinder. Without the removal of the solar drum, refurbishment would be difficult because of inaccessibility. The components will have to be modularized and externally located if orbital refurbishment is to be feasible. It is the manufacturer's opinion that component replacement is not the time consuming problem. Accommodating an unscheduled refurbishment cycle and testing time involved does, however, pose problems of some magnitude.

Table 5-3. Intelsat IV Parts List and Failure Rate Data

Unit, Description	Unit Weight kg	(lb)	Unit Failure Rate, Failures Per 10^6 Hr	Baseline	Weight Optimized
Communication					
Global beam receive antenna	2.21	4.87	0.01	1	1
Pre-amplifier chain	3.62	7.97	6.00	4	6
Transponder	10.48	23.10	not constant	12	12
Spot beam ant. & pos. mechanism	5.74	12.66	0.50	2	2
Antenna positioner electronics unit	0.64	1.42	0.50	1	3
Global beam transmitting antennas	3.98	8.77	0.01	1	1
CDPI					
Spinning & despun encoders, horn antenna & coupler	4.98	10.99	3.75	2	3
Omniantenna	4.43	9.77	0.01	1	1
Command receiver	1.53	3.38	1.20	2	3
Despun decoder	1.50	3.30	2.00	2	4
Spinning decoder	1.18	2.60	2.00	2	4
Attitude Control					
Propellant tanks; radial, spin-up & axial thrust chamber assemblies	8.34	18.39	2.90	2	4
Stabilization					
Sun & earth sensors, sensor selector & phase lock loop, torque generator, power supply demodulator filter	4.03	8.89	5.30	2	6
Power amplifier & motor windings	0.36	0.80	1.50	2	4
Bearing & power transfer assembly	25.01	55.14	0.50	1	1
Slip rings	0.07	0.15	0.20	2	3
Slip rings	0.07	0.15	0.20	2	3

Table 5-3. Intelsat IV Parts List and Failure Rate Data (Continued)

Unit, Description	Unit Weight kg	(lb)	Unit Failure Rate, Failures Per 10^6 Hr	Baseline	Weight Optimized
Electrical Power					
Main & charging solar arrays	36.00	79.36	0.20	1	1
Battery	19.63	43.28	0.10	1	2
Battery controller	2.05	4.52	0.50	1	3
Load relay	0.51	1.13	0.05	2	2
Main & charging solar arrays	36.00	79.36	0.20	1	1
Battery	19.63	43.28	0.10	1	2
Battery controller	2.05	4.52	0.50	1	3
Load relay	0.51	1.13	0.05	2	2

5.3.2 Tug Description

The Tug used in this study is the March 1972 MSFC baseline Tug (see Ref. 5.3), and the June 1972 Revision A Tug (Ref. 5.4). The overall size of the Tug is 4.9-m (14.67-ft) diam x 10.7-m (35-ft) length including the payload docking mechanism (see Figure 5-4). The nominal performance weights for round trip, retrieval only, and deploy only are:

Performance	Weight, kg (lb)
Deploy and Retrieval (round trip)	1,361 (3,000)
Retrieve	1,905 (4,200)
Deploy and Return Empty	3,673 (8,100)

These performance characteristics are based on a 296-km (160-nmi) x 28.5-deg inclination Shuttle parking orbit and payload injection into geosynchronous orbit (defined as synchronous equatorial in this document). The mass properties are:

Burnout weight	2,799 kg (6,173 lb)
Ignition weight, including 24,308 kg (53,600 lb) propellant	27,454 kg (60,538 lb)
Interface weight	663 kg (1,462 lb)
Gross weight (less payload	28,117 kg (62,000 lb)

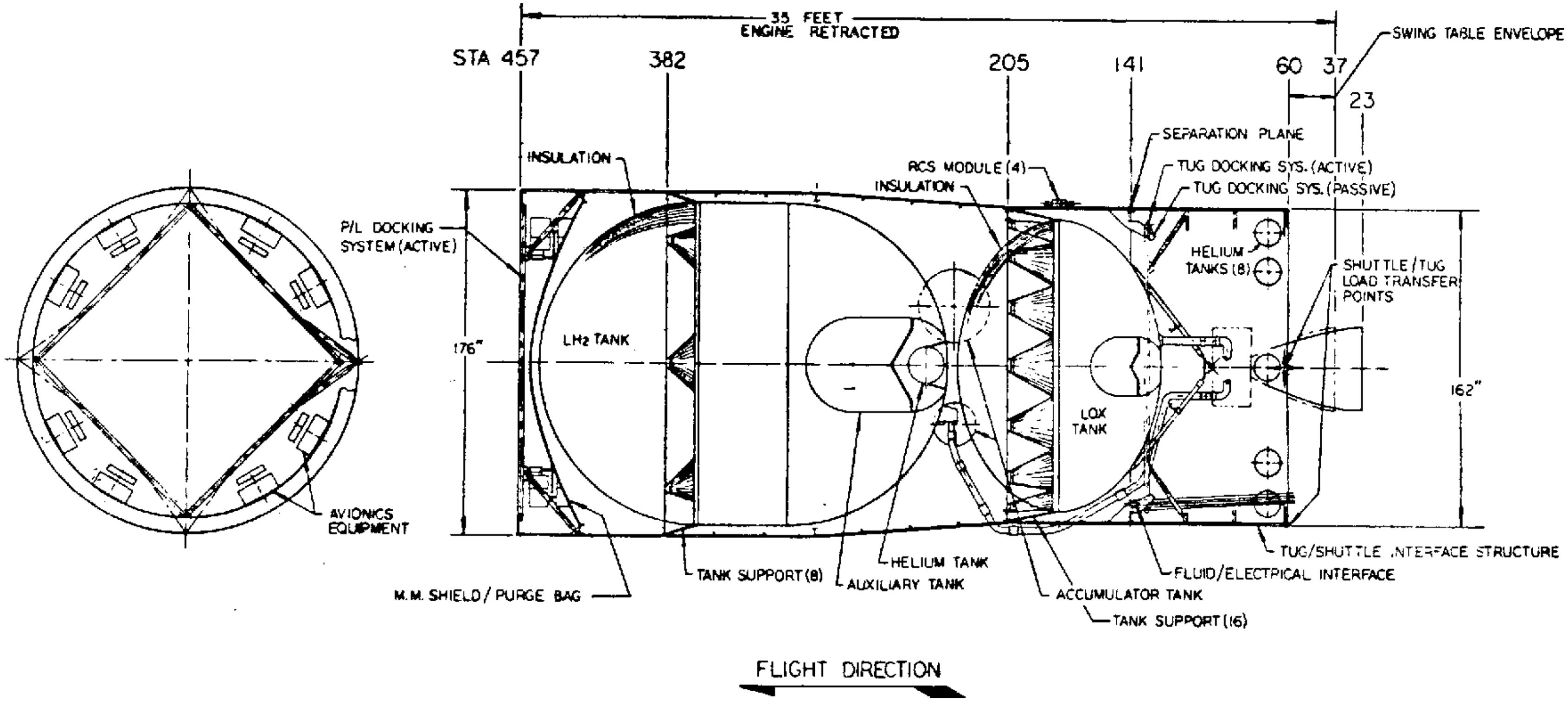


Figure 5-4. Baseline Tug Overall Configuration

The Tug can rendezvous and dock with a stable and passive payload that is adapted for retrieval. Remote man-in-the-loop TV for terminal docking is available as required in conjunction with rendezvous and docking laser radar.

5.3.3 Minimum Modification for Shuttle

This concept represents the minimum redesign that will be required to adapt the present Intelsat IV vehicle for Tug/Shuttle operation. Structurally the changes include replacing the apogee kick motor with a new thrust cone structure, which will serve as the main load-carrying structure, and mating the interface section with the Tug for launch and retrieval (see Figure 5-5). The kick motor can be eliminated since the Tug can place the satellite into synchronous equatorial orbit.

The expendable Intelsat IV satellite has been designed to take a high loading in an axial direction along its spin axis. If the mode of operation is to retrieve the satellite with the Tug and return it to the ground on the Shuttle for refurbishment, additional beef-up may be required to strengthen any marginal areas sufficiently to take the high lateral loading from reentry and landing.

A design study with similar objectives, but one conducted to identify detail design changes, was performed on the Defense Support Program satellite. This study resulted in a three-percent structural weight increase when the Shuttle/Tug was substituted for an expendable launch vehicle. The strengthen areas were limited to fittings, interface, and platform. The primary structure did not require beef-up.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 3

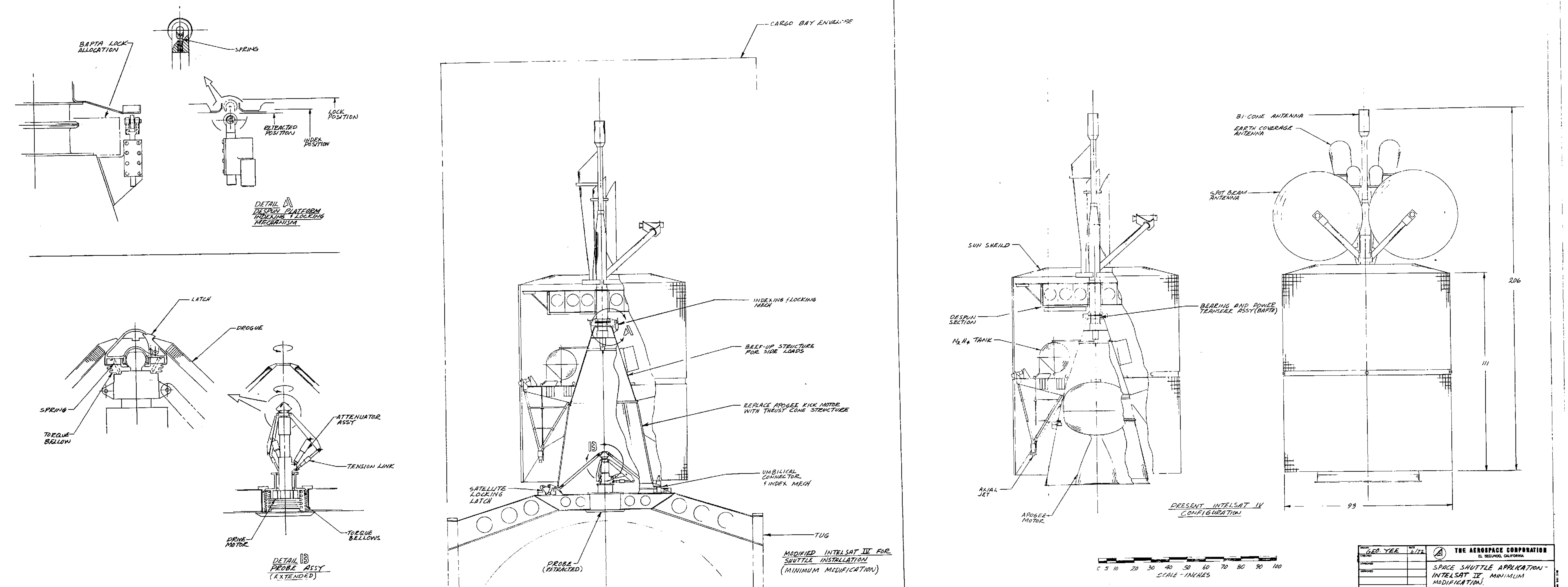


Figure 5-5. Present and Minimum Modification, Intelsat IV

DRAWN GEO. YEE	DATE 6/72	THE AEROSPACE CORPORATION, EL SEGUNDO, CALIFORNIA
CHECKED		SPACE SHUTTLE APPLICATION - INTELSAT IV, MINIMUM MODIFICATION.
APPROVED		SIZE / CODE IDENT NO. 12782 / DRAWING NO. ES-0155-026
APPROVED		SCALE NOTED / SHEET

The docking concept used in this study is depicted in Figure 5-5. It is the modified Apollo docking mechanism described in Ref. 5.4. It employs a basic Apollo drogue and probe mechanism which has been modified to enable the Tug-mounted probe to dock with the drogue on the spinning satellite. This is accomplished by pre-rotating the probe to match the 60-rpm spinning speed of the satellite prior to engagement. After engagement the probe motor is despun. The satellite is drawn up against a mating ring where it is indexed and latches are actuated to hard-dock the satellite to the Tug.

An umbilical connector attached to the satellite/Tug mating ring, when engaged, provides a power and command link from the Tug to the satellite and monitors the payload for safing and deactivation. At the time of this study the standard docking mechanism in the baseline Tug (Ref. 5.4) did not appear to be usable for retrieving a spinning satellite. Studies by McDonnell Douglas have since indicated, however, that spinning satellites can be retrieved if a circular ring is affixed to the payload, instead of the square ring specified in the baseline document. This approach appears to be less complex and more Tug/payload-adaptable.

The minimum modification Intelsat IV type of vehicle does not readily lend itself to an on-orbit service mode because of equipment inaccessibility due to the drum-mounted solar array arrangement. The two section drum structure is designed to slide off forward and aft to permit access to the equipment.

An area that should be examined in detail is the attitude control subsystem. This subsystem is most critical in the docking and despin operation and must be made compatible with the docking method. Studies have indicated that there are potential stability problems during payload despin after docking has occurred. The critical parameters are the relative stiffness, inertia, and spin rate. Moreover, docking with a spin-stabilized payload is not yet completely understood. These areas must be studied further and solutions for these potential problems found before retrieval of dual spin satellites can be given detail consideration. For this tradeoff study, however, it will be assumed that these technical problems can be solved and that retrieval is feasible.

5.3.4 Weight-Optimized Design

The outboard configuration of the weight-optimized design is identical to the minimum modification configuration for the Shuttle (Figure 5-5). The internal configuration, however, is modified to accommodate the increase in redundancy for the optimized design units. The increased redundancy was determined from the system optimization program (SYSOPT). The structural, or outboard, configuration was not changed because the structural reliability was assumed to be 100 percent for the life of the satellite. The increase in the units can be accommodated by an additional equipment platform within the solar drum.

SYSOPT is a FORTRAN computer program which optimizes the satellite system when weight or cost is the limiting factor. The measure in determining the optimum is MMD, which is the mean time to failure (MTTF) of the system in a finite time interval. A function of the system reliability, it is defined as:

$$\text{MMD}\ (T) = \int_0^T R(t)\, dt$$

where R(t) is the system reliability at time, t

This theoretical technique serves as a useful guideline to designers by allocating the weight (or cost) allotment between redundancies to improve system reliability and expendables and to extend system life in an optimal manner.

The input to SYSOPT is the baseline satellite system reliability mathematical model, failure rate data, and unit weights. The reliability model consists of a series of units which can be redundant. The system reliability is the product of the reliabilities of different types of units. A unit may contain internal redundancies such as active or standby redundancies and may contain several components and elements to make an assembly capable of independent operation. The failure rates of the units are assumed to be constant and do not consider burn-in or wearout failures.

The principle used in SYSOPT is an iterative process by which redundancies are added on the unit or sub-unit level. The increase in MMD over the increase in weight (or cost) is calculated for each unit. A unit where additional redundancies are not needed or are not feasible can be suppressed. The unit showing the greatest ΔMMD/Δ weight (or Δcost) is selected for redundancy, thereby formulating a new reference configuration. The process is repeated until the weight or cost constraint is reached. The process can also stop on a preselected number of iterations or ΔMMD/Δ weight ratio.

Since the weight is not to be considered a constraint for Shuttle payloads, the weight, cost, and number of iterations were removed as stops in the SYSOPT. The only computer stop instruction retained was the ΔMMD/Δ weight ratio which was set at 5×10^{-5} hrs/lb. On the basis of past experience this value was established as a practicable limit.

Those units which were suppressed from being redundant were:

(1) Global beam receive antennas
(2) Global beam transmit antennas
(3) Spot beam antenna
(4) Omni antenna
(5) Bearing and power transfer assembly
(6) Main and charging solar arrays
(7) Structure

The number of units were determined with those permitted to be redundant. The resulting weight-optimized equipment configuration is described in Table 5-2. The MMD and weight for the baseline and weight-optimized configuration are 6.1 years and 668 kg (1,473 lb), and 6.8 years and 778 kg (1,715 lb), respectively. The MMD was based on a truncation time (amount of expendables) of nine years. The weights do not include 739 kg (1,630 lb) for the apogee motor.

5.3.5 Redesign Configuration

This redesign concept has been developed to make use of the large diameter available in the Shuttle by expanding the solar array drum diameter to 4.6 m (15 ft) and shortening the drum length. This configuration, shown in Figure 5-6, maintains the same power and spin-despun stabilization method. The decrease in length will provide space for other payloads to share the transportation cost. The large diameter will minimize the tendency of the vehicle to tumble during any despun abnormalities. The large diameter also offers the possibility of utilizing larger spot beam antennas to increase communication capability, or to rearrange the antennas to decrease the overall payload length.

The larger despun diameter also makes it feasible to locate the weight-optimized equipments externally above the spinning drum envelope. This provision permits easy access for assembly, testing, and equipment changeout without the need to remove the solar array drum. The optimized equipment list can be packaged into modules to simplify the refurbishment for ground or on-orbit operations. Furthermore, these modules and expendables when mounted on the despun platform will not be sensitive to the mass distribution, in contrast to mounting them on the spinning portion.

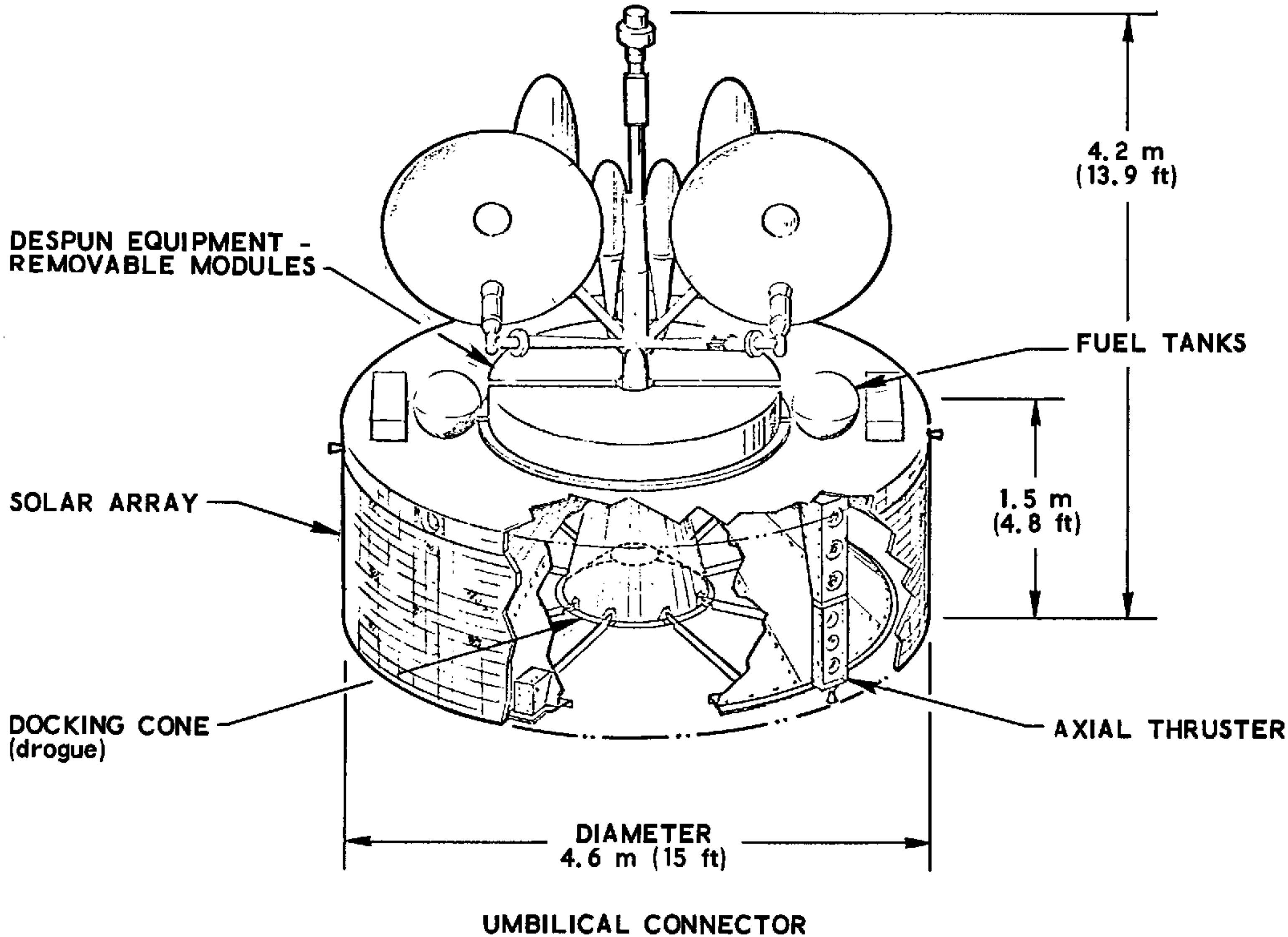


Figure 5-6. Redesigned Configuration, Intelsat IV Task

The overall length of the redesign configuration is approximately 4.2 m (13.9 ft) which compares with 5.3 m (17.5 ft) for the baseline. This length consists of 1.4 m (4.75 ft) for the solar array drum, 0.5 m (1.65 ft) for the despun equipment module, and 2.3 m (7.5 ft) for the antenna assembly. The drum length was established by the 4.6-m (15-ft) diameter cylinder and by maintaining the same electrical power level as the baseline. The antenna arrangement shown is the baseline dimension. The antenna assembly could be rearranged to effect an overall length established by the diameter of the large spot beam antennas. This would result in a net payload length of approximately 3.5 m (11.5 ft) which would be short enough to tandem two satellites on the Tug and fit in the Shuttle cargo bay. In such an arrangement the total payload weight would be within the Tug performance in the deployment mode but would be in excess of the Tug round-trip capability. The tandem arrangement could be employed only for the initial deployment mode or multiple with other payloads.

5.3.6 Weight Estimate

The weight statements for the baseline reusable, weight-optimized, and redesign configurations were estimated. They are shown in Table 5-4. The baseline expendable weights were obtained from a contractor document (Ref. 5.2). The baseline reusable uses the equipment from the baseline expendable except for the following modifications required to deploy and retrieve the satellite by the Tug.

Table 5-4. Intelsat IV Weight Statements

Configuration / Subsystem	Baseline Expendable Config A		Minimum Modification for Shuttle, Config B		Weight Optimized Config C		Redesign Config D	
	kg	(lb)	kg	(lb)	kg	(lb)	kg	(lb)
Structure	184	(405)	270	(595)	284	(625)	238	(525)
Basic	143	(315)	143	(315)	(143)	(315)	136	(300)
Supports	41	(90)	41	(90)	(54)	(120)	54	(120)
Retrieval		--	86	(190)	(86)	(190)	48	(105)
Guidance and Navigation	34	(75)	34	(75)	(51)	(113)	51	(113)
Propulsion (apogee kick motor)	739	(1630)		Tug	Tug			Tug
Attitude Control	152	(334)	175	(386)	421	(927)	395	(871)
Equipment	7	(16)	7	(16)	15	(32)	15	(32)
Propellant	135	(297)	157	(346)	379	(836)	356	(784)
Tanks	10	(21)	11	(24)	27	(59)	25	(55)
Telemetry and Command	23	(50)	23	(50)	35	(77)	35	(77)
Electrical Power	118	(259)	118	(259)	172	(380)	172	(380)
Mission Equipment (Communications)	159	(350)	159	(350)	167	(368)	167	(368)
Payload Weight	1408	(3103)	778	(1,715)	1,129	(2,490)	1,059	(2,334)
Tug Modifications			170	(375)	170	(375)	170	(375)
Sub Total	1408	(3103)	948	(2,090)	1,300	(2,865)	1,229	(2,709)
Tug		--	28,123	(62,000)	28,123	(62,000)	28,123	(62,000)
Total Weight	1408	(3103)	29,071	(64,090)	29,422	(64,865)	29,352	(64,709)

Modification		Weight Change
1.	Structurally beef-up docking gear, and add equipment for Tug retrieval.	+ 86 kg (190 lb)
2.	Remove apogee motor since Tug places the satellite in orbit.	-739 kg (1,630 lb)
3.	Increase attitude control weight to maintain the same velocity increment (tank and propellant).	+ 24 kg (52 lb)
	Satellite net weight change	-629 kg (1,388 lb)
4.	Structurally modify the Tug to enable it to retrieve the spinning satellite.	+170 kg (375 lb)
	Gross weight change	-459 kg (1,013 lb)

The baseline reusable gross weight is 778 kg (1,715 lb) and the total weight chargeable to the payload is 948 kg (2,090 lb).

The mass of the weight-optimized concept was estimated by using the baseline reusable weight plus the increases for system optimization. The changes for the baseline reusable are:

Modification		Weight Change
1.	Structure supports for the unit increases	+13.6 kg (30 lb)
2.	Guidance navigation for increase in redundancy	+17.2 kg (38 lb)
3.	Attitude control for longer life and weight increases	+245.3 kg (541 lb)
4.	CDPI for increase in redundancy	+12.2 kg (27 lb)
5.	Electrical power for increase in redundancy	+54.9 kg (121 lb)
6.	Mission equipment for increase in redundancy	+8.2 kg (18 lb)
	Satellite net weight change	+351.4 kg (775 lb)

The levels of redundancy are shown in Table 5-3 for each of the subsystems. The gross optimized weight is 1,129 kg (2,490 lb) and total weight chargeable to the Tug is 1,299 kg (2,865 lb).

The redesigned configuration represents a concept which has the weight-optimized equipment externally accessible and the overall length shortened to multiple the payloads. In shortening the solar array drum, the thrust cone length was reduced and the structural beef-up of the baseline structure for landing loads was eliminated because of the increased strength with the short structure. The equipments that are mounted on the despun platform are not weighted for modularization. The weight reflects only accessibility. The structural weight for modularity is estimated to increase by 90 percent for those subsystems that are modularized (Ref. 5.5). The total weight would increase by approximately 305 kg (670 lb), or 0.90 x 399 kg (880 lb) minus 54 kg (120 lb) of current supports. If this increase is added to the estimated weight without modularization of 1,229 kg (2,709 lb) the total satellite weight is increased to 1,533 kg (3,381 lb).

5.4 DISCUSSION

The spin-despun Intelsat IV was examined as a representative automated spacecraft for Shuttle/Tug operation. This examination consisted of a minimum modification, and weight-optimized and shortened redesign configurations. The minimum modification study indicated that the design change needs to be relatively minor for Tug compatibility. The weight change for the minimum modification was a reduction of approximately 454 kg (1,000 lb). The reduction is due to the elimination of the 739-kg (1,630-lb) apogee motor. The gross weight including structural add on to the Tug is 948 kg (2,090 lb). This weight is within the 1,361 kg (3,000 lb) round-trip performance of the Tug.

The weight-optimized configuration maintains the baseline reusable configuration but increases the subsystem equipments to increase the satellite life from 6.1 to 6.8 yr MMD. The increase in the equipments is based on optimizing the unit redundancies. This increase in the number of units can be accommodated within the configuration by adding a platform. The gross weight of this concept is 1,299 kg (2,865 lb) and is within the Tug round-trip performance capability.

The shortened redesign configuration is an attempt to utilize the available diameter of the Shuttle cargo bay for purposes of launching two payloads in tandem and to provide equipment accessibility for assembly, test, and component replacement. It was determined that increasing the solar array drum diameter to 4.6 m (15 ft) decreased the overall length by 0.9 m (3 ft). If the antenna assembly were rearranged, the overall length could be reduced by an additional 0.9 m (3 ft) resulting in an overall length of 3.5 m (11.5 ft). Given this length, it appears feasible to tandem two of these configurations on one Shuttle/Tug flight; the gross payload weight for two satellites, however, will exceed the Tug round-trip capability. Two satellites in this arrangement can be launched in the deployment mode only. If on-orbit maintenance is required, the satellite must be modularlized and the modules located in the despun section to minimize the balancing problems. Modularity could increase the total weight to approximately 1,542 kg (3,400 lb) plus spare modules and remote manipulator weights. This total weight is in excess of 1,361-kg (3,000-lb) round-trip and can exceed the 1,905-kg (4,200-lb) Tug retrieval performance capability.

5.5 REFERENCES

5.1 Intelsat IV Third Quarterly Progress and Status Report, SSD 90230R, 1 April through 30 June 1969, Hughes Aircraft Company, Space Systems Division (July 1969).

5.2 Unpublished Hughes Weight Statement of the Intelsat IV, Y-1 Qualification Model for Flight Model Design.

5.3 Intelsat IV System Summary, Hughes Aircraft Company (August 1968).

5.4 Satellite Description for OOS Follow-on Satellite Retrieval Study, 72-2810A-027 (ATM), M. G. Wolfe, The Aerospace Corporation (21 April 1972).

5.5 Payload Effects Analysis Study, LMSC-A990556, Final Report, Lockheed Missiles and Space Company (30 June 1971).

5.6 Integrated Operations/Payloads/Fleet Analysis Phase II Second Interim Report, ATR-71(7231)-11, Vol II, NASA Payload Data, The Aerospace Corporation (March 1971).

6. Observatory
Spacecraft

6. OBSERVATORY SPACECRAFT

6.1 INTRODUCTION

For the study of observatory spacecraft HEAO-C was selected as the representative payload for this class of satellites. In the 1972 NASA mission model the HEAO-C is scheduled for launch in 1979. It will continue to function until 1983, when additional HEAOs with different mission equipment are scheduled to be launched (see Figure 6-1). Prior to mission C, HEAO A and B will have been flown in 1975 and 1977. Because of this continuing program HEAO-C requirements (e.g., pointing accuracy) are being included in the A&B design to ensure that the same spacecraft design will be used. The mission objectives for HEAO flights starting in 1983 are not defined at this time; it is assumed, however, that these flights will not duplicate prior missions. Each HEAO mission will be unique.

One concept using the same spacecraft throughout the 12 years in the Shuttle era was continued in this tradeoff study. The basic structural configuration and subsystems were established to accommodate the A, B, and C mission equipments. In selecting alternate concepts the design study emphasized the modifications necessary for compatibility with the Shuttle to deploy, retrieve, and perform on-orbit service and ground refurbishment. This study also investigates the equipment modifications required to shorten and extend the design life about the nominal two-year MMD. The study was conducted to provide payload data on various design lives for the economic tradeoff of optimum life for various revisit times.

6.2 MISSION OBJECTIVE AND EQUIPMENT DESCRIPTION

The overall objective of HEAO-C is to investigate detailed structure, spectra, and location of specific X-ray sources using pointed (1 arc min) spacecraft. (See Vol. II of this report.) HEAO-A&B, which preceed mission C, are to perform a scanning survey of the celestial sphere primarily to locate the X-ray sources, whereas HEAO-C will be devoted entirely to pointing in order to obtain data on the structure, spectra, and polarization of the source.

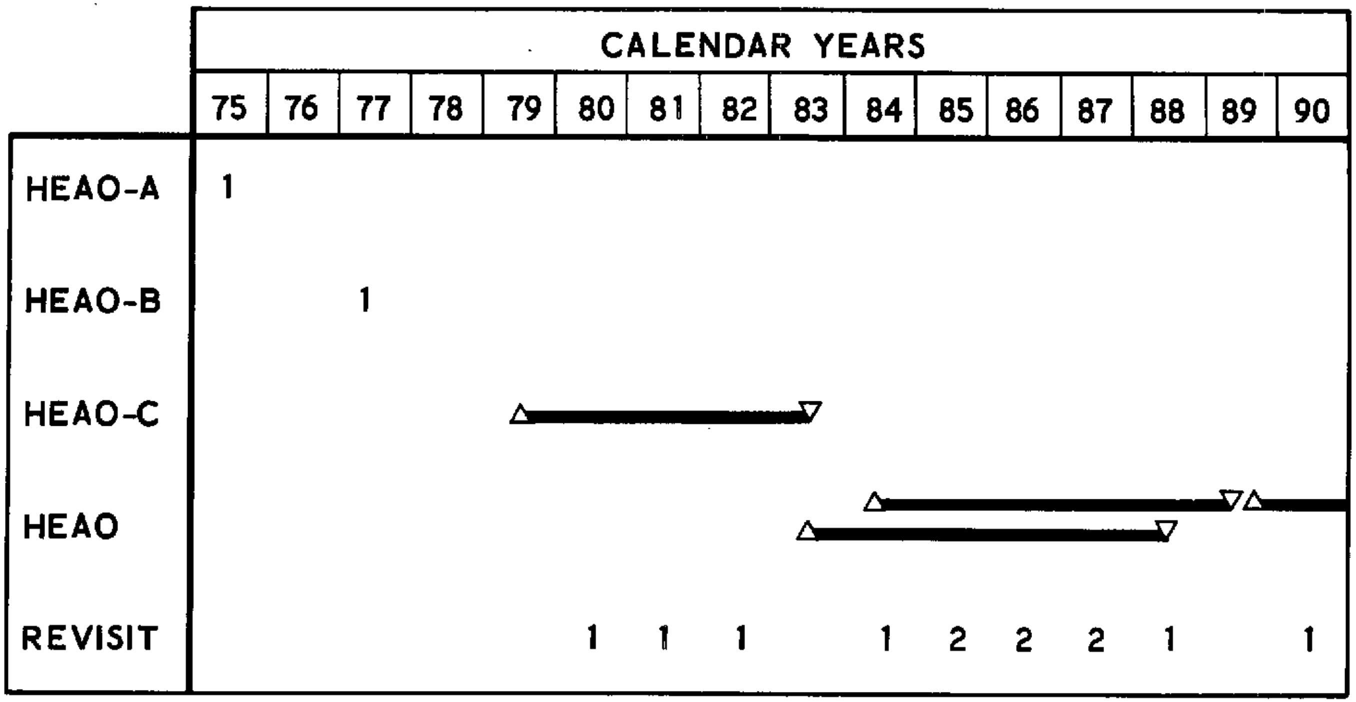


Figure 6-1. Traffic Model

The experiments are listed in Ref. 6.1 and are duplicated in Table 6-1 for the convenience of the reader. The structural configuration for HEAO-A&B was designed to accommodate the HEAO-C mission experiments. In addition to accommodating these experiments, the stabilization and control includes the mission C pointing requirement of ± 1 arc min. The HEAO-A&B spacecraft design is basically adaptable for HEAO-C i.e., high commonality with the primary difference being limited to the experiments (Ref. 6.2). This configuration is shown in Figure 6-2 with solar array deployed and the orbit adjust stage mounted for expendable launch on T-III D. The overall length is 13.1m (42.9 ft) and the distance across the octagon is 2.7m (8.8 ft).

6.3 DESIGN

The objectives of this design study were to provide design information for an economic comparison of ground refurbishment, on-orbit remote service and man-tended service, and spacecraft design life. To develop and study feasible approaches, conceptual designs were performed on the configurations summarized in Table 6-2. The ground refurbishment concepts are considered minimum modification since only payload support, deployment, retrieval, and safety monitoring are involved in the Shuttle compatibility, whereas on-orbit service requires all new spacecraft structure and packaging for performing maintenance on-orbit. The on-orbit service by remote manipulator was investigated for three Shuttle docking locations. The shirtsleeve man-tended service considered only the forward docking position.

The HEAO baseline design is two-year MMD, and it was varied from one to five years' design life. The design life variation was performed by varying the equipment redundancies. The structural, thermal control, and mission equipment were not involved in this reliability analysis.

Table 6-1. Experiment Power, Weight, Size, and Field of View

Experiment Components	Field of View (Total Angle) (min)	Size H × W × L (in.)	Total Weight (lb)	Average Power (W)	
				Inside Tube	Outside Tube
High Resolution Telescope					
HR Mirror Assembly	60	40.5 dia × 48 long	2025		
HR Image Intensifier		10 dia × 10 long	20	13.5	
Mode 1	17 × 17				
Mode 2	2.1 × 2.1				
Mode 3	17 × 4.2				
Electronics Module		5 × 5 × 10	24		19.0
HR Crystal Spectrometer	60	In transport mechanism	397	21.0[b]	
HR Experiment Transport Mechanism		38 × 33 × 83	70	(c)	
Objective Grating	(a)	40 dia × 12 long	60	(c)	
Filter Wheel	(a)	10 dia × 6 long	40	(c)	
Structure			Excluded		
Large Area Telescope					
LA Mirror Assembly	60	40 dia × 49.5 long	2400		
LA Position Sensitive Detector	26	5 × 5 × 5	100	1.0	
Electronics Module		5 × 5 × 10	50	13.5	
Solid State Detector	7	10 dia × 18 long plus 18 in. dia sphere	90	2.0	
Electronics Module		7 × 7 × 7	25		6.2[b]
LA Experiment Transport Mechanism		38.5 × 40 × 21	115	(c)	
Filter Wheel	(a)	10 dia × 6 long	40	(c)	
Structure			Excluded		
Flare Detectors					
Coarse Detector (6)	90 × 90 (ea.)	5.5 × 8 × 3 (ea.)	120		12
Fine Detector (1)	8	10 × 10 × 48	27		(d)
Electronics Module (3)		5 × 5 × 5 (ea.)	150		28 (2 only)
Monitor Proportional Counter	1 FW[e] ½ FWHM[f]	10 × 16 × 23	98		1.0
Electronics Module		Included above	44		8.0
Flat Crystal Spectrometer (1)	1 FW ½ FWHM	25 × 19 × 63	162		17.0
Aspect Detector	7.5	10 dia × 18 long	100		8.0
Electronics Modules		5 × 5 × 10	21		6.0
Cabling			157		
On Tubes			27		
On Spacecraft			130		
Low Energy Telescope				24	
Mirror Assembly	1.5	20 dia × 26 long	400		
Position Sensitive Detector	1.5	5 × 5 × 5	30		
Low Background Detector	1.5	5 × 5 × 6	30		
Filter Wheel		10 dia × 6 long	30		
Transport Mechanism		Included above	50		(g)
Structure			300		

a. Depends on detector.
b. Not to be included in maximum power configuration.
c. Uses 6 W during operation; duty cycle is negligible.
d. Fine detector and its electronics use 2 W and 8 W, respectively, during operation; used only during flare observation.
e. Full width.
f. Full width, half maximum.
g. Uses 6 W during operation; duty cycle is negligible.

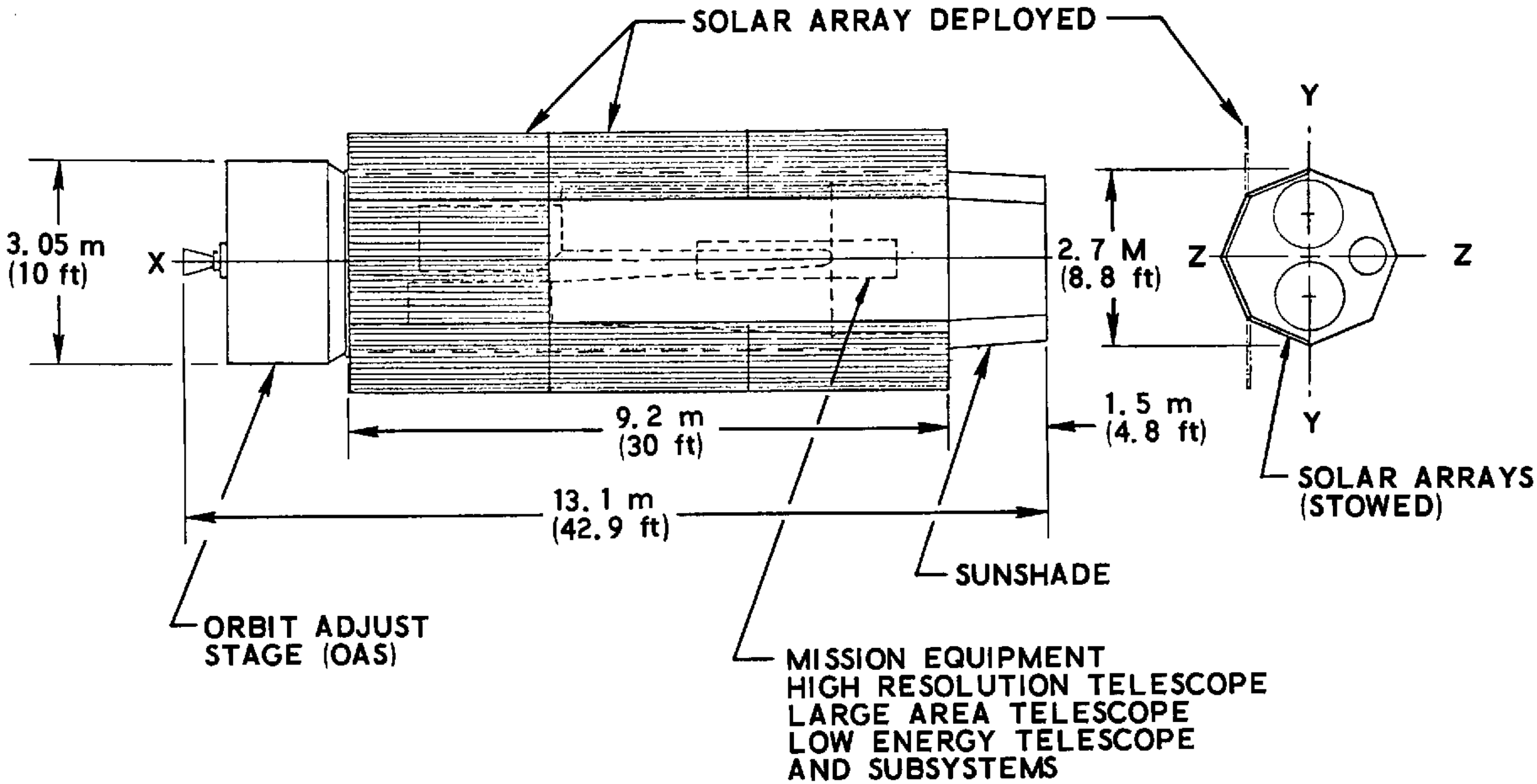
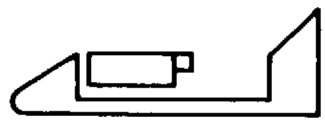
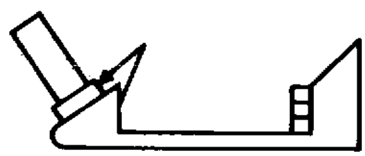
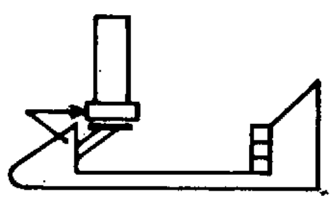
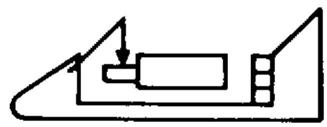
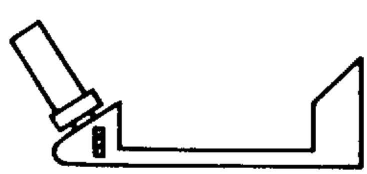


Figure 6-2. Expendable Configuration - HEAO C

Table 6-2. Summary of Configurations

CODE	TITLE	CONFIGURATION	REMARKS
A	GROUND REFURB MODIFIED BASELINE		MINIMUM MODIFICATION REFURB. ON GROUND
B	ALTERNATE BASELINE		RETRACTABLE SUN SHADE
C	ON ORBIT - REMOTE SERVICE NOSE DOCKING		SERVICE WITH MANIPULATOR SPARES INCLUDED
E	TILT TABLE DOCKING		
F	CARGO BAY DOCKING		REPAIR & C/O IN STOWED POSITION
D	MAN-TENDED IVA ON-ORBIT SERVICE NOSE DOCKING		SERVICE BY IVA

6.3.1 Modified Baseline

The modified baseline is the minimum modification version of the expendable satellite for deployment and retrieval by the orbiter. This concept is shown in Figure 6-3 without the orbit adjust stage (OAS) since the orbiter can place 27,210 kg (60,000 lb) into 463-km (250-nmi)/28.5-deg circular orbit with supplemental OMS tankage installed. A similar configuration is shown in Figure 6-4 where the telescope sunshade is postulated to be retractable, thereby providing added space in the cargo bay to permit installing a second HEAO or other satellite of comparable size.

In both configurations, there is sufficient cross-sectional area in the cargo bay for the solar arrays to be fixed in their extended positions, thus obviating the need for an array extension mechanism. The internal equipments are located to provide accessibility for ground assembly, checkout, and component replacement. The spacecraft subsystem equipments are isolated on a panel located on the anti-solar side. The equipment boxes are an open rack concept for easy access. Each equipment is thermally integrated and self-contained. This approach to accessibility and passive thermal control is also consistent with Shuttle ground refurbishment concepts.

For cargo bay stowage both concepts require a pallet which can be the standard payload pallet to transfer the HEAO loads to hard points in the orbiter. The manipulator is assumed to be the deployment and retrieval mechanism. With the pallet and manipulator concept, the payload docking provisions are hard points for the manipulator to grasp; the pallet transfers the loads rather than the docking mechanism.

The HEAO is shown in the forward position, which is within the orbiter center-of-gravity limits; however, the more preferred center-of-gravity location is aft, which leaves the forward position for other payloads that may require more orbiter/payload interface. It appears that HEAO has minimum interfacing and can therefore be located aft. The space available for other payloads with configurations A and B is 6.7 and 9.2m (22 and 30 ft).

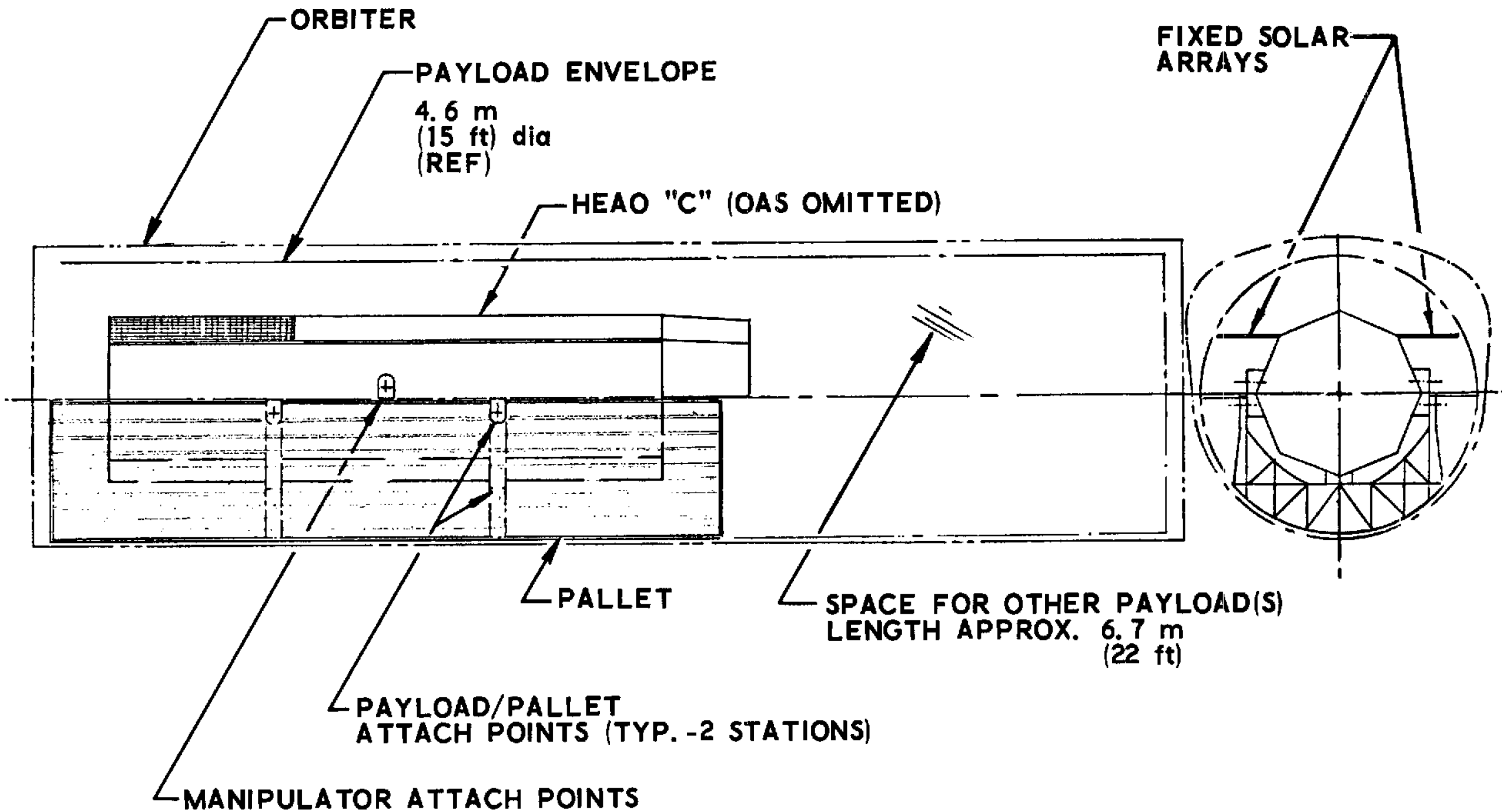


Figure 6-3. Configuration A - Modified Baseline

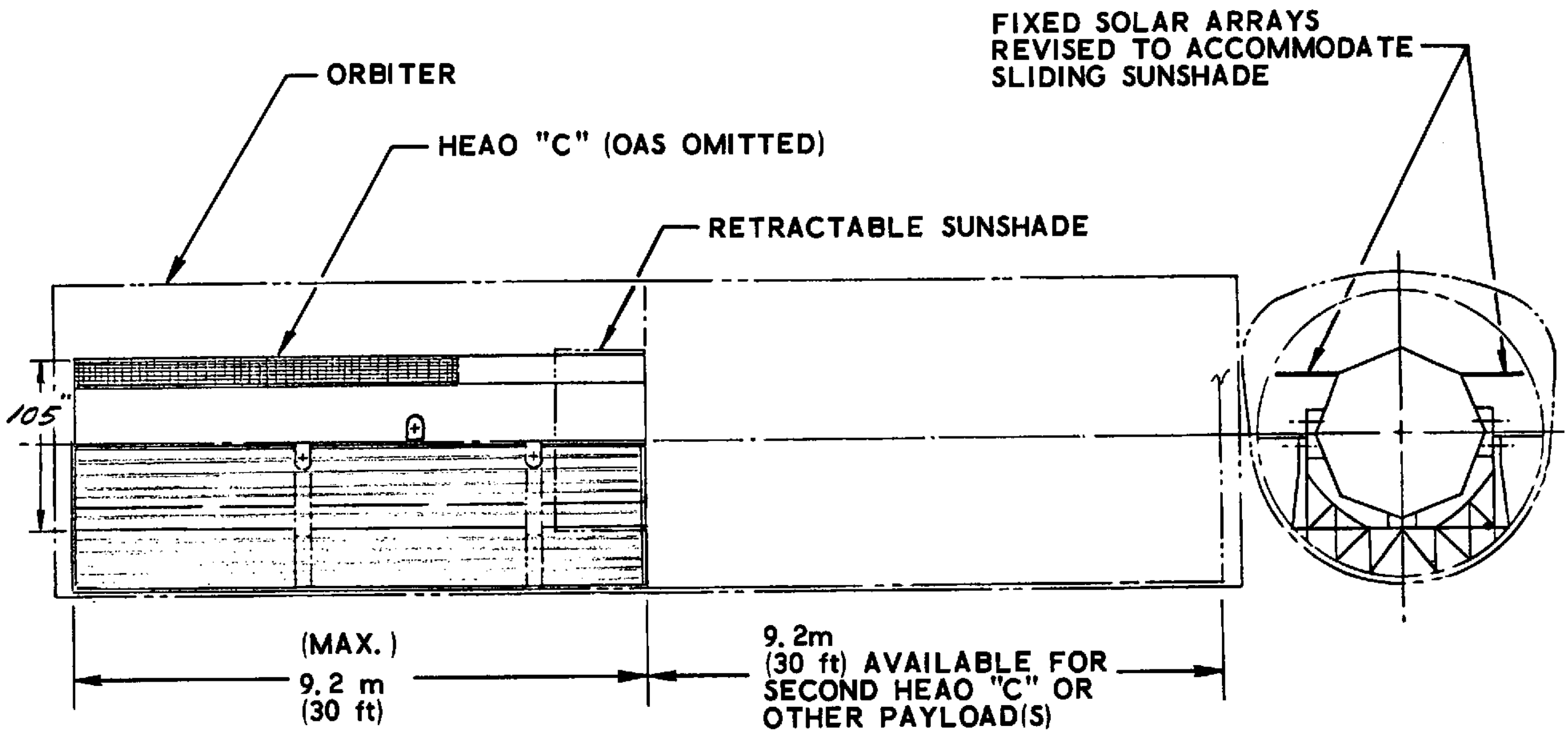


Figure 6-4. Configuration B - Alternate Modified Baseline

6.3.2 On-Orbit Remote Service

The concepts for remote on-orbit service are shown in Figures 6-5, 6-6, and 6-7. These concepts consist of nose docking, tilt table docking, and cargo bay docking in which the manipulator articulation and operator viewing capability vary. For predeployment testing, the manipulator must transfer the payload from the cargo bay to the nose docked position to perform the initial payload checkout. The nose docking can use the existing orbiter docking mechanism and is in good pilot view for payload retrieval since the terminal docking maneuvers can be performed without the use of the manipulator.

The component replacement is achieved in the three cases by the orbiter equiped manipulator's removal of failed modules and replacing them with spare modules. The spare modules are located on racks in the cargo bay, where the replaced (failed) modules also can be stored. EVA is available as a backup mode for the nose docking and tilt table docking. For the nose docked position, the manipulator reach to replace modules is afar and only modules located in direct viewing positions are accessible. The payload must be rotatable for access to all the modules.

The tilt table method, shown in Figure 6-6, should reduce the manipulator reach and improve the operator line of sight. The viewing distance for the module replacement is closer, and if the arms can articulate as shown the modules in the blind area can be replaced. The viewing in these areas can be augmented with small TV zoom cameras and lights located on the arm probe. An alternative approach is to adapt the docking mechanism to rotate and index the payload into position for direct module removal and replacement.

Figure 6-7 shows the HEAO remaining in the cargo bay for predeployment checkout and on-orbit service. The spacecraft is rectangular, permitting unidirectional module extraction with the payload in the cargo bay. In this approach all of the modules are accessible from the top side by the

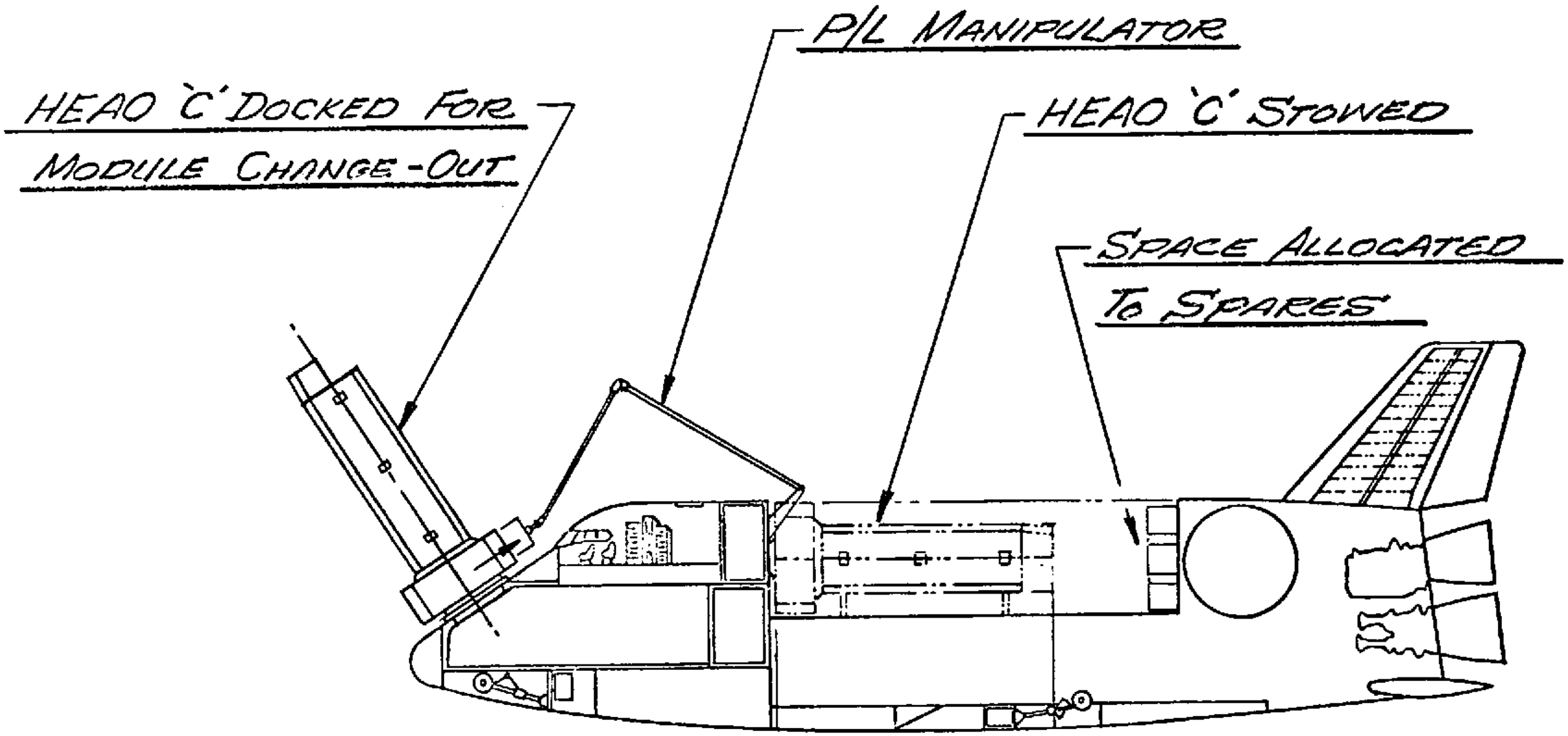


Figure 6-5. Configuration C, On-Orbit Service, Nose Docking

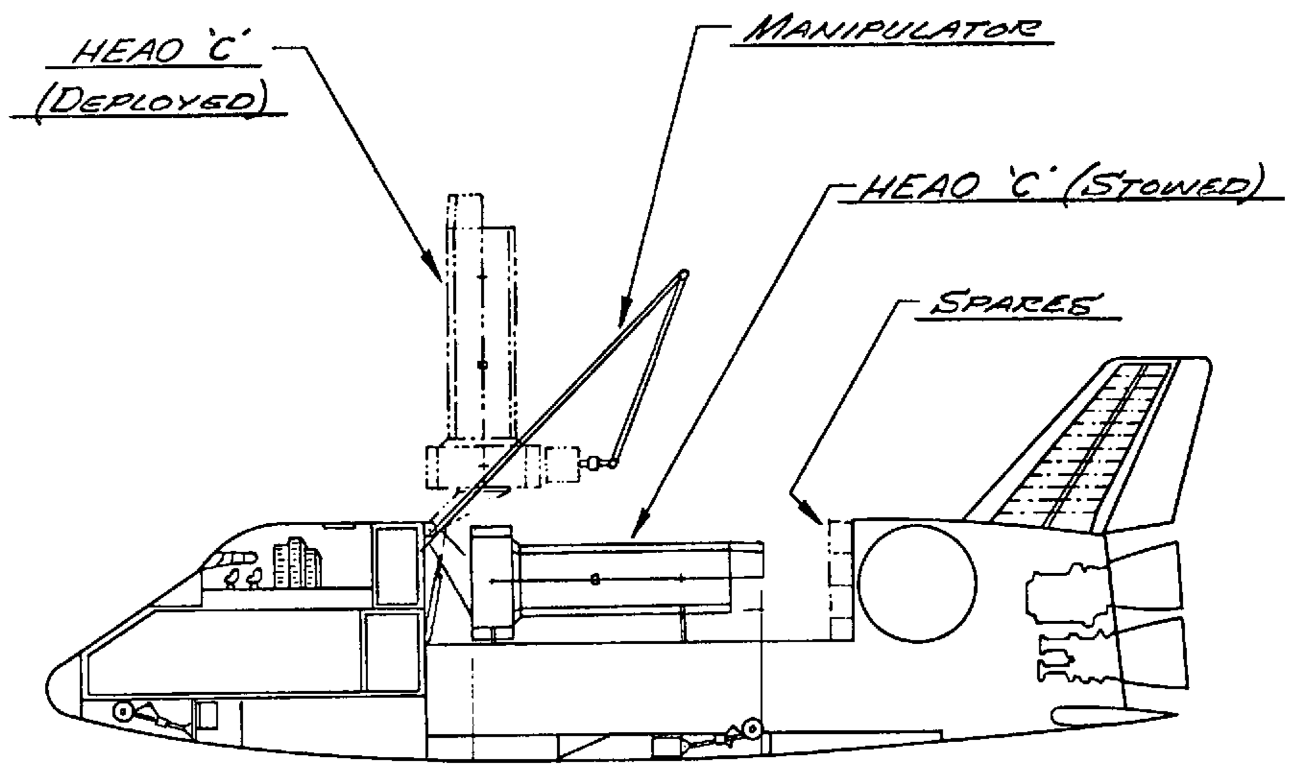


Figure 6-6. Configuration E, On-Orbit Service, Tilt Table

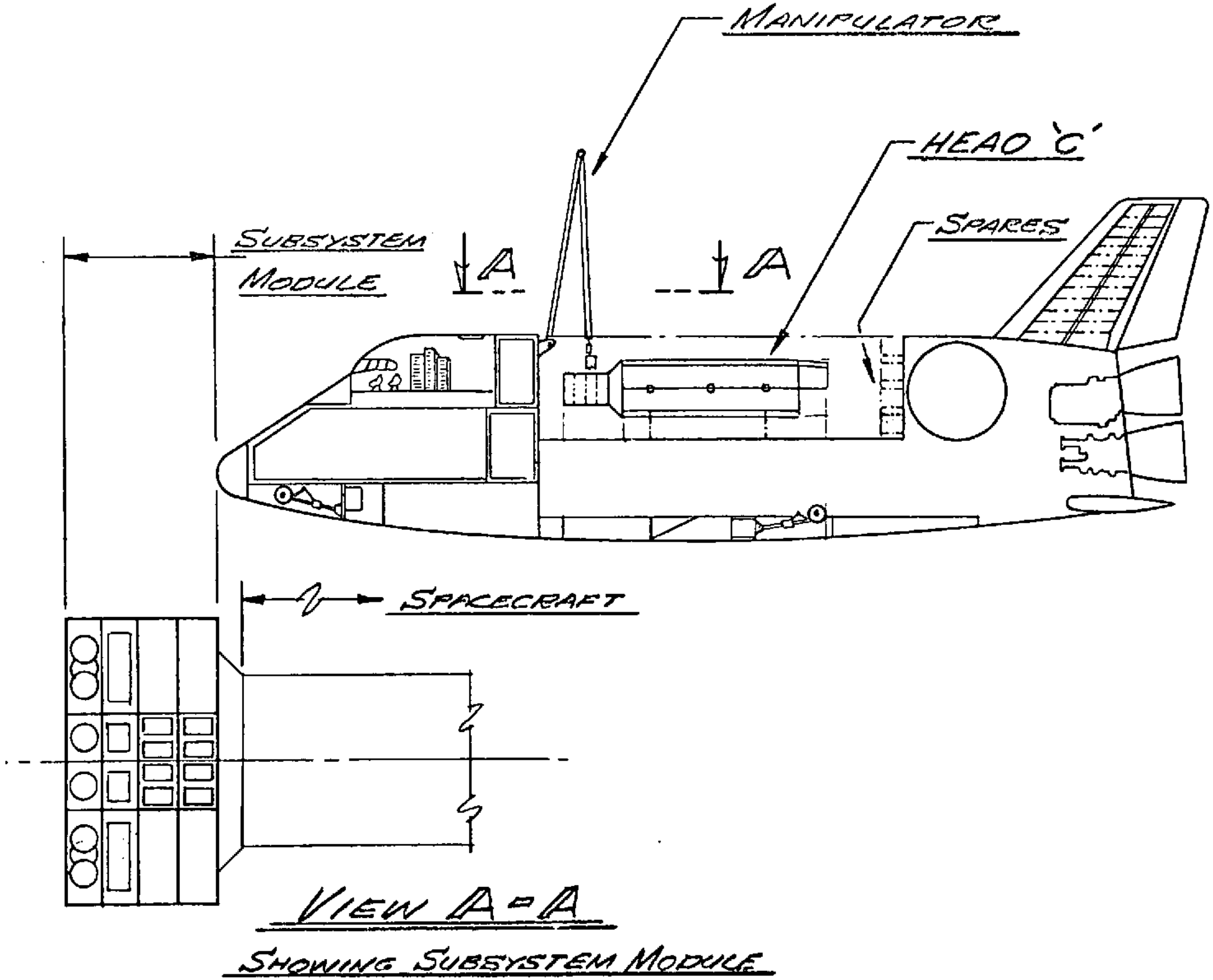


Figure 6-7. Configuration F, On-Orbit Service, Cargo Bay Docking

manipulator or IVA as a backup mode with direct operator line of sight. This concept also simplifies ground access to payload subsystems from post mate to launch. The arrangement has disadvantages, however, in that the available cargo space is not used efficiently and the on-orbit checkout is performed in the bay where the instruments and guidance sensors have no direct space view to provide end-to-end checks. A better arrangement is to mount the unidirectional access configuration on the tilt table where accessibility and sensor solar sighting can be attained with good viewing, simplification of manipulator operation, and good deployment approach.

The layout of the components in the modules for on-orbit replacement is shown in Figure 6-9. The component modularization groups the items by failure rates and subsystem. The higher failure rates within the same subsystem are mounted in the same module in accordance with area availability. The failure rates in descending order are listed in Figure 6-9 from highest down to five percent failure in two years. This list is approximately 50 percent of the master equipment list which appears in Vol. I of Ref. 6.1. The 50 percent represents 70 percent of the weight in the master list and requires 16 of the 24 modules. The eight spare moudles can be used for the higher reliability items and those instruments in the mission equipment that can be modularized. The module sizes are approximately 0.6m (2 ft) high x 1.2m (4 ft) wide x 0.6m (2 ft) deep. The battery module is the heaviest item. Its weight, not including the structural box, is 76 kg (168 lb).

The 4.6m (15 ft) octagon spacecraft configuration was selected to utilize the large available volume in the cargo bay and to provide accessibility and growth of the mission sensors. The space for growth is available by additional layers of modules.

The modularization for the unidirectional access concept is shown in Figure 6-10. This arrangement represents the same total volume with half the number of modules used in the large octagon configuration. The reaction control subsystems, located at the four corners to provide the greatest

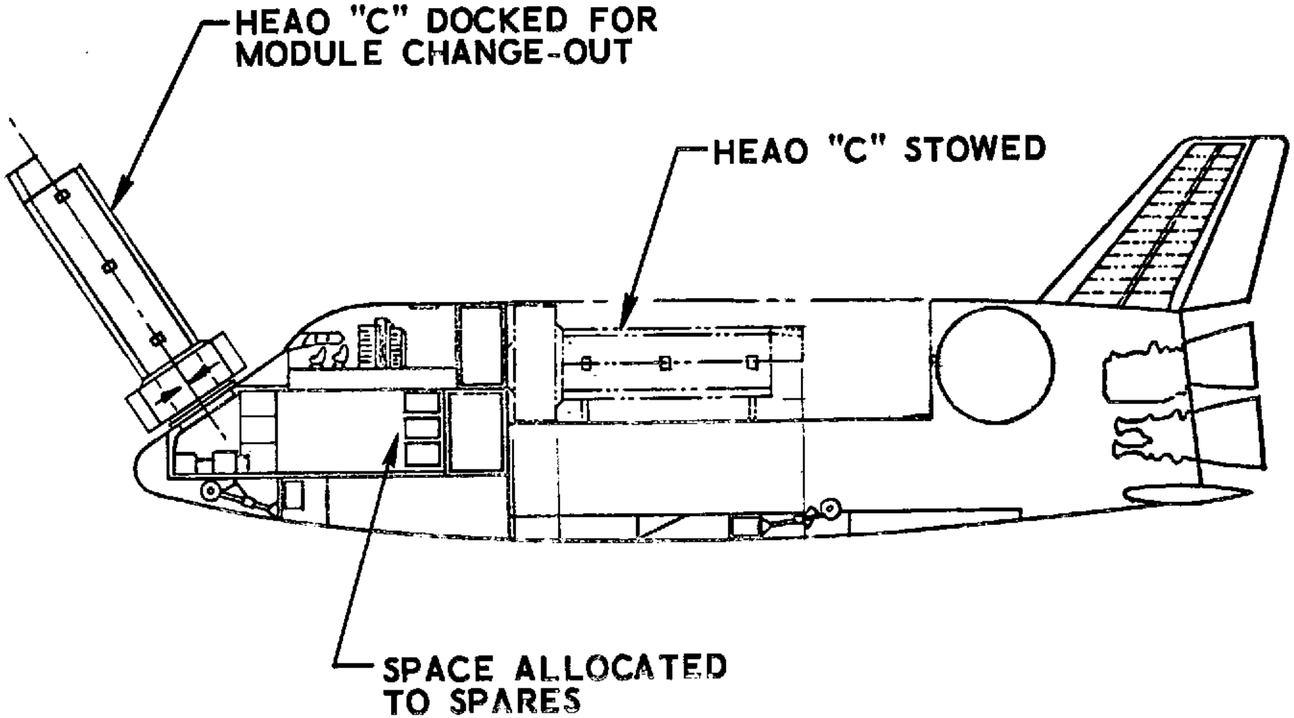


Figure 6-8. Configuration D - Man-Tended, Nose Docking

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distances between thrusters, are self-contained units except for the electrical connections. The unidirectional module access simplifies ground assembly, testing, and replacement because the units are all located on one side, minimizing the need for special ground handling tools. This concept can also accommodate equipment growth by adding modules.

6.3.3 Man-Tended

The man-tended version is illustrated in Figure 6-8 as a nose docking arrangement since an airlock is assumed to be available only at this docking position. The spacecraft is a pressurizable structure with meteoroid protection and a crew transferable docking mechanism for IVA shirtsleeve environment. A large cylindrical structure [4.6m (15 ft)] was selected to provide adequate room for personnel mobility in repairing and testing components and for mounting subsystem components along the structural shell for direct accessibility.

The initial pressurization is provided by the payload, but the attendant environmental control is assumed to be provided by the orbiter through the 1-m hatch opening. With the shirtsleeve IVA condition provided in the payload, the modularity concept should be a simpler design than those that require automation or EVA to replace equipments. The technician could use his ground testing and equipment replacement experience in many of the on-orbit component replacement operations. Repair training for payload ground refurbishment should be applicable to on-orbit refurbishment. The repair techniques should be analagous except for the zero gravity effects.

This method, however, will require identification of failed components and a high degree of self-check capability. The system test can be performed by the orbiter-provided payload checkout system. The on-orbit repair time is limited to seven days Shuttle-on-station time since other orbiter functions will need to be performed. The payload must be designed to have this rapid repair capability by identifying and isolating the failed component, simplifying component replacement operation, and automatically performing system test. In addition to facilitating spacecraft repair, the IVA approach can also resupply, adjust, recalibrate, and repair the mission equipment as required if it is accessible.

6.3.4 Spacecraft Design Life

The baseline HEAO-C is an expendable payload with a two-year design life. For a reusable payload the two-year design life may not be optimum and the design life should be varied to determine the optimum life. To modify the payload design life in the most rigorous manner, payload failure rate data and a reliability model are required. This type of information was not available, only overall reliability data being available in Vol. III of Ref. 6.1.

An approximate approach was used in performing this task with only overall reliability data. This approximation has been proven empirically on past satellite programs to adequately characterize the overall reliability for preliminary design studies. The reliability equation is:

$$R(t) = e^{(-At-Bt^2)}$$

where

A = coefficient for no redundancy (single string)

B = coefficient for redundancy

t = time

The coefficient can be determined by reliability values at two different times per satellite configuration since there are only two unknowns. The reliability values were provided in the reference document at one and two years for three payload configurations. Also, the reliabilities for two-year design with increasing redundancy, based on optimizing the reliability increase per cost increase, were available and are tabulated in Table 6-3. From this set of data the coefficients were estimated and the MMDs were computed for a two-year truncation time.

Table 6-3. Reliability for Increasing Redundancy (2-yr Lifetime)

Added Component	Spacecraft Total Quantity	Component Cost ($)	(2 yr) Spacecraft Reliability	Mean Mission Duration
1. Baseline Spacecraft Total (2 yr)			0.6433	1.757
2. Read Only Memory	2	1 000	0.6541	1.765
3. Data Storage Control	2	30 000	0.6953	1.814
4. Electrical Integration Assembly	9	2 000	0.6972	1.816
5. Memory	3	20 000	0.7085	1.827
6. Receiver	5	37 500	0.7284	1.847
7. Command Processor	3	20 000	0.7380	1.857
8. Regulator	5	30 000	0.7506	1.868
9. CMG Electronics	5	172 000	0.8318	1.902
10. Solar Panel Deployment Mechanism	7	10 000	0.8362	1.904
11. Remote Decoder	13	8 000	0.8396	1.905
12. Processor/Computer	4	99 000	0.8658	1.923
13. Battery Charger	7	100 800	0.8872	1.932
14. Electrical Integration Assembly	10	2 000	0.8877	1.934
15. Transfer Assembly	2	99 000	0.9061	1.944
16. DSS Electronics	3	52 000	0.9153	1.950
17. Rate Gyros Set (3 Gyros)	3	60 000	0.9260	1.957
18. PCM Encoder	3	60 000	0.9359	1.962
19. Thruster Modules	5	57 000	0.9453	1.968
20. Read Only Memory	3	1 000	0.9454	1.968
21. CMG Electronics	6	172 000	0.9634	1.978
22. Remote Decoder	14	8 000	0.9642	1.978
23. Format Generator	3	40 000	0.9676	1.980
24. Power Control Assembly	3	25 000	0.9697	1.983
25. Regulator	6	30 000	0.9719	1.984
26. Data Storage Control	3	30 000	0.9738	1.987
27. Cabling Conversion Set	3	2 000	0.9739	1.987
28. Processor/Computer	5	99 000	0.9792	1.988
29. Clock	2	26 000	0.9806	1.989
30. Memory	4	20 000	0.9816	1.990
31. Tube Insulation	2	4 000	0.9818	1.991
32. Coatings	2	4 500	0.9819	1.991
33. Outer Shell Insulation	2	4 500	0.9821	1.991
34. Battery Charger	8	100 800	0.9860	1.992
35. Command Processor	4	20 000	0.9867	1.993
36. Electrical Integration Assembly	11	2 000	0.9868	1.993
37. CMG Electronics	7	172 000	0.9895	1.994
38. Remote Decoder	15	8 000	0.9896	1.995
39. Star Tracker Electronics	5	150 000	0.9913	1.996

These coefficients were used also in determining the reliabilities at four and five years' design life or HEAO at various levels of redundancy. The computed reliability and MMD are tabulated in Table 6-4.

From this tabulation the redundancy level was selected for the reliability at design life. The baseline expendable HEAO has a 0.6433 spacecraft reliability at two-year design life. Table 6-4 reveals that the nearest reliabilities at the four- and five-year design point are for configuration number 17 and 20, respectively. The corresponding MMDs are 3.5 and 4.3 years.

This same payload was also reduced in reliability, since the Shuttle is expected to perform yearly revisits. The reduction is reliability was performed by removing redundancy in the order shown in Table 6-5. The redundancy order from the top was based on a decreasing weight to reliability ratio.

The same reliability at design life was used again to select the level of redundancy. Eliminating eight components reduced the reliability to 0.629 at one year.

The MMD at this redundancy level is 1.3 years for a two-year truncation time. The MMD is larger than the design life in this case because the truncation time is twice the design life, whereas in the extended life the design life and truncation time were the same.

These levels of redundancies, as developed by the above process, were used to define the equipment list for the one-, four-, and five-year HEAO-C. The structure, thermal control, and mission equipment were not changed from the baseline configuration. This assumption should be satisfactory for the structure and thermal control since the redundancies are in standby mode and adequate space appears to be available for the added redundancy. Extending the life of the mission equipment is questionable, however, because the data value, instrument deterioration, technology, and calibration are functions of time. These factors do not affect the reliabilities in this study since the reference data did not include mission equipment failure rates.

Table 6-4. Increase Reliability for Four and Five-Year Design Life

No.	Added Component	Spacecraft Total Quantity	Component Cost ($K)	(4-yr) Spacecraft Reliability	Mean Mission Duration (4-yr mission)	(5-yr) Spacecraft Reliability	Mean Mission Duration (5-yr mission)
1	Baseline Spacecraft Total (2 yr)			.0865	2.312	.0200	2.359
2	Read Only Memory	2	1.0	.0948	2.346	.0231	2.398
3	Data Storage Control	2	30.0	.1300	2.500	.0375	2.576
4	Electrical Integration Assembly	9	2.0	.1319	2.507	.0383	2.585
5	Memory	3	20.0	.1437	2.550	.0438	2.635
6	Receiver	5	37.5	.1664	2.527	.0551	2.730
7	Command Processor	3	20.0	.1822	2.646	.0685	2.732
8	Regulator	5	30.0	.1952	2.717	.0707	2.842
9	CMG Electronics	5	172.0	.3550	3.042	.1887	3.309
10	Solar Panel Deployment Mechanism	7	10.0	.3405	3.020	.1757	3.272
11	Remote Decoder	13	8.0	.3749	3.077	.2063	3.363
12	Processor/Computer	4	99.0	.4571	3.144	.2093	3.578
13	Battery Charger	7	100.8	.4775	3.266	.3034	3.653
14	Electrical Integration Assembly	10	2.0	.5127	3.317	.3415	3.742
15	Transfer Assembly	2	99.0	.5760	3.416	.4123	3.909
16	DSS Electronics	3	52.0	.6094	3.468	.4512	3.997
17	Rate Gyros Set (3 gyros)	3	60.0	.6503	3.529	.5010	4.105
18	PCM Encoder	3	60.0	.6901	3.588	.5510	4.208
19	Thruster Modules	5	57.0	.7296	3.644	.6025	4.311
20	Read Only Memory	3	1.0	.7300	3.645	.6028	4.312
21	CMG Electronics	6	172.0	.8123	3.756	.7166	4.522
22	Remote Decoder	14	8.0	.8162	3.761	.7222	4.531
23	Format Generator	3	40.0	.8323	3.783	.7451	4.573
24	Power Control Assembly	3	25.0	.8415	3.797	.7576	4.598
25	Regulator	6	30.0	.8516	3.8122	.7719	4.625
26	Data Storage Control	3	30.0	.8600	3.825	.7831	4.647
27	Cabling Conversion Set	3	2.0	.8601	3.826	.7832	4.649
28	Processor/Computer	5	99.0	.8884	3.860	.8264	4.718
29	Clock	2	26.0	.8955	3.869	.8370	4.736
30	Memory	4	20.0	.9003	3.876	.8439	4.749

Table 6-5. Reduced Reliability

No.	Name	No. Changed From	To	Cum ΔW	No. ΔW	Rel (1 yr)
1	Digital Sun Sensor	2	1	0.75	0.75	0.9237
2	Fixed Head Star Tracker	4	3	23.75	23	0.8977
3	Batteries	6	5	79.75	56	0.8367
4	Control Power Unit	2	1	95.75	16	0.8094
5	Chargers	6	5	109.75	14	0.7771
6	Tape Recorders	4	3	124.75	15	0.7218
7	Electrical Integ Assy	10	9	126.75	1.5	0.7125
8	Transponder*	2	1	149.55	93.3	0.6293
9	Wide Angle Sun Sensor Elect	3	2	150.55	1	0.6256
10	CMG and Elect	4	3	166.75	16.2	0.5712
11	Wide Angle Sun Sensor	3	2	167.75	1	0.5678
12	PSK Demodulation	2	1	169.75	2	0.5591
13	Multiplexer	2	1	171.75	2	0.5481
14	Format Gen	2	1	175.75	4	0.525
15	Regulator	4	3	183.75	8	0.4798
16	Digital Process	3	2	203.75	20	0.3685
17	Command Memory	2	1	209.75	6	0.3355
18	PMC Encoder	2	1	213.75	4	0.3114
19	Digital Sun Sensor Elect	2	1	215.75	2	0.2899
20	Command Process	2	1	217.75	2	0.2670

*2 receivers and 1 transmitter (each)

In the extended life case the life limited items were examined to determine the reason for the limitation and to develop methods of eliminating or reducing the limits. The results of this examination are summarized in Table 6-6 and indicate that the life limited item can be extended to four and five years.

In certain areas, redundancy is difficult to implement. This study did not attempt to investigate the technical feasibility of attaining the level of redundancies, i.e., gyro operational laws for four gyros are available but no fully satisfactory methods have been found for five or more gyros.

6.3.5 Weight Estimate

The weight estimates for the HEAO-C, shown in Table 6-7, were developed from the expendable weights in Ref. 6-1. The weights for the modified baseline configuration were estimated to reflect the conceptual design changes required to adapt the expendable HEAO-C to the orbiter as reusable HEAO-C. These changes are:

(1) Add docking mechanism and hard points for deployment and retrieval.

(2) Delete orbit adjust stage.

(3) Add standard pallet for payload stowage.

(4) Retract sunshade for configuration B.

The major changes for the external docking configuration (C&E), which includes nose docking and tilt table docking, are:

(1) Add docking mechanism and hard points for deployment and retrieval.

(2) Delete orbit adjust stage.

(3) Add simple payload supports for stowage.

Table 6-6. Life Limited Items

	Life Limited Item	Reason	Methods of Eliminating Reducing
1	RCS thrusters	Not space qualified even for two years	No problem anticipated
2	Fuel tank bladder	Not space qualified even for two years	Correct materials must be selected
3	Tape recorders	Wearout	40,000 hr possible at present cycle rate; extra recorders could be added if required
4	Batteries	Sized for two years	Extra dormant batteries could be added, as required
5	Solar cells	Sized for two years	No problem, add more cells
6	Gas (expendables)	Sized for two years	Add more gas or go to magnetic control
7	Counter gases*	Sized for two years	Add more gases
8	Liquid helium* (if required)	1-year supply	These experiments will be completed in first year
9	X-ray optics*	Deterioration with evaporation	Close temperature control, high quality of gases (some degradation possible)
10	Proportional counters*	Disposition on wires	Close temperature control, high quality of gases (some degradation possible)
11	Detectors*	Color centers fade with light; lower sensitivity in latter stages of mission	Plan critical experiments early
12	Thin windows*	May become brittle electron bombardment	Plan critical experiments early
13	Moving parts* (bearing, turret, etc.)	Wear	Redesign problem since no close tolerances required (i.e., bearing noise should not be a problem)

*Experiment area (from discussions with Dr. A.B.C. Walker)

Table 6-7. HEAO-C Weight Estimates, Design Configuration Variation

Configuration / Subsystem	Baseline Expendable		A/B Modified Baseline		C/E Ext Dock		D Man-Tended		F Cargo Bay Docking	
	kg	(lb)	kg	(lb)	kg	(lb)	kg	(lb)	kg	(lb)
Structure	1797	(3,961)	1865	(4,111)	3271	(7,212)	4208	(9,278)	2648	(5,837)
Basic	620	(1,367)	620	(1,367)	620	(1,367)	620	(1,367)	620	(1,367)
Equipment Section		(0)		(0)	535	(1,180)	829	(1,829)		(0)
Meteorite Protection		(0)		(0)		(0)	506	(1,115)		(0)
Telemetry Supports	888	(1,958)	888	(1,958)	888	(1,958)	888	(1,958)	888	(1,958)
Module Supports*		(0)		(0)	531	(1,170)*	531	(1,170)*	531	(1,170)*
ACS Structure	170	(374)	170	(374)	533	(1,175)*	607	(1,339)*	490	(1,080)*
Adapter	96	(212)	96	(212)	96	(212)	96	(212)	96	(212)
Docking Mechanism		(0)	45	(100)	45	(100)	108	(238)		(0)
Sun Shade	23	(50)	45	(100)	23	(50)	23	(50)	23	(50)
Environmental Control	229	(505)	229	(505)	229	(505)	359	(791)	229	(505)
Equipment	229	(505)	229	(505)	229	(505)	229	(505)	229	(505)
Manned		(0)		(0)		(0)	130	(286)		(0)
Guidance and Navigation	396	(872)	396	(872)	396	(872)	396	(872)	396	(872)
Propulsion Apogee Motor	948	(2,089)		(0)		(0)		(0)		(0)
Attitude Control	479	(1,055)	479	(1,055)	592	(1,305)	675	(1,488)	544	(1,199)
Propellant	385	(848)	385	(848)	476	(1,049)	543	(1,196)	437	(964)
Inerts	94	(207)	94	(207)	116	(256)	132	(292)	107	(235)
Telemetry & Command	81	(178)	81	(178)	81	(178)	81	(178)	81	(178)
Electrical Power	445	(981)	445	(981)	445	(981)	445	(981)	445	(981)
Mission Equipment	3255	(7,175)	3255	(7,175)	3255	(7,175)	3255	(7,175)	3255	(7,175)
Payload Weight	7628	(16,816)	6748	(14,877)	8268	(18,228)	9418	(20,763)	7596	(16,747)
Pallet		(0)	1242	(2,738)		(0)		(0)		(0)
Payload Bay Supports		(0)		(0)	413	(911)	471	(1,038)	380	(837)
Tilt Table					113	(250)				
Total Weight	7628	(16,816)	7990	(17,615)	8795	(19,389)	9889	(21,801)	7976	(17,584)

*includes modularization

(4) Add large cylindrical spacecraft structure to house the subsystem equipment and to provide adequate internal room for IVA access to mission equipment.

(5) Modularize the relocated subsystem equipment into modular drawers for manipulator handling.

(6) Add propellants to maintain the same attitude control velocity.

The cargo bay docking configuration (F) has the same estimated weight as the external docking concepts except that the docking mechanism and the large cylindrical structure are eliminated. These features were eliminated to provide unidirectional access to the modules while positioned in the cargo bay.

The man-tended configuration requires additional hardware to permit module remove and replace in a shirtsleeve IVA environment. Such hardware includes a habitable module to accommodate the subsystem equipments and work area, crew and module transferable docking mechanism, and crew atmosphere. Environmental control was assumed to be provided by the orbiter through the hatch opening. The modularity weights were assumed to be the same even though the man-tended modules would be less complex than the remote refurbishment techniques because the equipment need not be as encapsulated. The weight of the 4.6-m (15-ft) diam x 1.8-m (6-ft) long habitable module was based on the unit weight factors described in section 4.3.7. As expected, the man-tended version has the largest total weight, 9.886 kg (21,800 lb), which is within the orbiter/payload center-of-gravity limits.

The weight estimate for design life variation is shown in Table 6-8 for one, four, and five years. The structural configuration used in these estimates is the baseline configuration. The subsystem weights were based on the equipment list to be found in section 6.3.4, above.

Table 6-8. HEAO-C Weight Estimates, Design Life Variation, Configuration A/B

Subsystem \ Configuration	1-yr life		2-yr life		4-yr life		5-yr life	
	kg	Wt (lb)	kg	Wt (lb)	kg	Wt (lb)	kg	Wt (lb)
Structure	1808	(3,986)	1865	(4,111)	2023	(4,460)	2103	(4,637)
Basic	620	(1,367)	620	(1,367)	620	(1,367)	620	(1,367)
Supports	888	(1,958)	888	(1,958)	888	(1,958)	888	(1,958)
ACS Str. (0.354 x ACS)	113	(249)	170	(374)	328	(723)	408	(900)
Sun Shade	45	(100)	45	(100)	45	(100)	45	(100)
Adapter	45	(100)	45	(100)	45	(100)	45	(100)
Docking Mechanism	96	(212)	96	(212)	96	(212)	96	(212)
Environment Control	229	(505)	229	(505)	229	(505)	229	(505)
Guidance and Navigation	385	(848)	396	(872)	416	(917)	416	(917)
Apogee Motor	-	-	-	-	-	-	-	-
Attitude Control	255	(562)	479	(1,055)	926	(2,041)	1154	(2,543)
Propellant	192	(424)	385	(848)	769	(1,696)	962	(2,120)
Inerts	31	(69)	31	(69)	31	(69)	35	(78)
Tank (0.163 x W_p)	31	(69)	63	(138)	125	(276)	157	(345)
Telemetry and Communications	64	(140)	81	(178)	105	(231)	107	(236)
Electrical	405	(892)	445	(981)	457	(1,007)	457	(1,007)
Mission Equipment	3255	(7,175)	3255	(7,175)	3255	(7,175)	3255	(7,175)
Payload Weight	6399	(14,108)	6748	(14,877)	7410	(16,336)	3255	(17,020)
Pallet	1242	(2,738)	1242	(2,738)	1242	(2,738)	1242	(2,738)
Total Weight	7641	(16,846)	7990	(17,615)	8652	(19,074)	8962	(19,758)

6.4 OPERATIONS

The operational concepts for the various configurations were briefly examined and compared in order to operationally rank the designs. They are grouped into interface, ground operations, and on-orbit operations. Those areas that were considered equal in each design concept were not included, i.e., EMI has equal task complexity for all configurations. The areas considered were based on those areas which have operational differences between design concepts.

In performing this evaluation several assumptions concerning the orbiter capability were made, including:

(1) The manipulator is capable of deploying a payload the size of HEAO (9,070 kg/20,000 lb) from the cargo bay and redocking to a docking mechanism, or holding the payload in space to the orbiter stabilization capability.

(2) The manipulator has sufficient dexterity to remotely remove 22 to 181-kg (50 to 400-lb) modules and replace them without damage to spacecraft or module.

(3) The manipulator can remotely operate manually operable knobs and levers.

(4) Standard payload checkout set can duplicate the ground system test automatically and isolate the failure.

The ranking in each area was performed by assigning (+) = 1 (advantageous), (0) = 0 (neutral), or (-) = -1 (disadvantageous). The assignment of these values was relative in that it was a comparison of concepts. It was not based on current expendable launch approaches. The selected areas and assigned values are listed in Table 6-9. There was no weighting applied to account for some areas having greater economic or design impact.

Table 6-9. HEAO-C Operations Assessment

Area	A/B Modified Baseline	C Nose Docking	E Tilt Table Docking	F Cargo Bay Docking	D Man-Tended Nose Docking
Interface					
Stowage hardware	(0) St'd pallet	(+) P/L provided	(+) P/L provided	(+) P/L provided	(+) P/L provided
Umbilicals	(+) One fixed	(-) Two fixed	(0) One deploy	(+) One fixed	(-) Two fixed
Docking mechanism	(+) None	(0) One	(0) One	(+) None	(-) Crew trans
Deployment mechanism	(+) Manipulator	(+) Manipulator	(-) Tilt table	(+) Manipulator	(+) Manipulator
Cargo volume use	(+) Good use	(-) Lg P/L + Spare	(-) Lg P/L + Spare	(-) Waste space	(-) Lg P/L
Subtotal	4	0	-1	3	-1
Ground Operations					
Ground testing	(0) Non-modular	(0) Special tool	(0) Special tool	(+) Accessible	(0) Special tool
Deployment tests	(+) None required	(-) Not possible	(0) Tilt table	(+) None required	(-) Not possible
Post-mate access	(-) No access	(0) Partial access	(0) Partial access	(+) Accessible	(+) Accessible
Subtotal	0	-1	0	3	0
On-orbit Operations					
Test preparation	(+) Cargo bay	(-) Transfer & umb	(0) Erect	(+) Cargo bay	(-) Transfer & umb
Manipulator ops	(+) Deploy	(-) Deploy & refurb	(0) Refurbishment	(-) Deploy & refurb	(+) Deploy
Pre-deployment C/O	(-) No space view	(+) Space view	(+) Space view	(-) No space view	(+) Space view
Operator view for refurbishment	(0) No refurb	(-) Partial view	(0) Close but partial	(+) Close in	(0) None required
Deploy extendables	(-) In cargo bay	(+) Manip deploy	(+) Manip deploy	(-) In cargo bay	(+) Manip deploy
Refurbishment method	(+) Ground	(-) Remote	(-) Remote	(-) Remote	(0) Shirtsleeves IVA
Backup repair mode	(+) Ground	(0) EVA	(0) EVA	(+) IVA	(+) IVA
Unplanned repair	(+) Ground	(-) EVA	(-) EVA	(0) IVA	(0) IVA
Subtotal	3	-3	0	-1	3
Total	7	-4	-1	5	2

Note: (+) = 1, (0) = 0, (-) = -1

The interface area considered only the mechanical aspects, since power, data rates, communication, etc., were assumed to be the same for all concepts. The summation of the interface values indicate that the modified baseline and cargo bay docking have the cleanest interface. In the ground operations area the cargo bay docking is favored because of the good accessibility provided by the unidirectional access modules. For on-orbit service, the modified baseline and the man-tended operations appear to be the desirable concepts. The modified baseline is good because of the simple deployment operation. The man-tended method is desirable since the shirtsleeve IVA permits performance of many of the operational functions, and the manipulator is limited to payload deployment and deployment of extendables.

6.5 SUMMARY

Three conceptual designs for refurbishing the HEAO-C were developed in the cost tradeoff study. The ground refurbishment required satellite modifications to provide compatibility with the cargo bay, deployment, and retrieval interfaces. The on-orbit servicing by remote manipulator and man-tended methods required, in addition to the ground refurbishment requirement, designs compatible with the orbiter manipulator arms and shirtsleeve IVA approach to removing and replacing modules.

These concepts were then functionally examined for interface, ground operations, and on-orbit operations.

By assuming equal merit for the three groups, the modified baseline and cargo bay docking are the desirable operational approach. If, however, the on-orbit operation is given more importance, the man-tended version should be included as a candidate concept for further evaluation. The nose docking concept should not be given further consideration.

The feasibility of on-orbit servicing by remote manipulator is dependent on the capability of the orbiter manipulator arms to provide servicing operations. It is therefore recommended that NASA include in the orbiter manipulation requirements that they have the capability to remove and replace modules and operate knobs and levers. It is also recommended that NASA follow through with HEAO-C spacecraft design/cost study efforts on a two-year life HEAO-C spacecraft by modifying the HEAO-B spacecraft design for Shuttle compatibility, retrieval, and HEAO-C experiments. Emphasis should be on experiment integration, experiment service on-orbit, adaptation of the HEAO-A&B experiment and spacecraft structure to HEAO-C, and reliability of the mission equipment.

6.6 REFERENCES

6.1 High Energy Astronomy Observatory Mission C, Phase A, Final Report, NASA TMX-64652, Vol I, II, III, NASA/MSFC (January 1972).

6.2 HEAO, Design Approach Summary, TRW Systems Group (27 August 1971).

7. Cost Analysis

7. COST ANALYSIS

7.1 INTRODUCTION

Cost analyses were performed for the following three payload programs:

- Solar Observatory Program with Shuttle Sortie
- System Demonstration Program consisting of Tracking and Data Relay Satellite and System Test Satellite
- Observatory Program consisting of High Energy Astronomical Observatory Satellite

The analysis involved preparation of inputs, and generation of costs estimates and cost comparisons to display tradeoffs. Results of the tradeoffs were used to select low life cycle cost design and operational approaches, and to assist in developing program concepts.

Basic procedures used in developing relative cost values were (1) preparation of payload design data for cost analysis, (2) definition of program concepts, (3) determination of program schedules, and (4) operation of the payload cost model. Preparation of payload design data involves weight and subsystem characteristics information. Program concepts involve the type of program, payload characteristics, and operational mode variations required for the trade studies. Capture analysis established the schedules for each payload launch vehicle type assigned. Schedules included those for launch, retrieve, refurbish, maintenance, and redesign. In order to reduce launch costs the payload mission model was searched for payloads which can share the trip. The costing tool used to provide cost estimates is Aerospace's Payload Cost Model (PALCM). Cost information provided by the model consists of program direct costs, which are the sum of the payload program and the direct launch costs. The payload program costs are broken out into phases for RDT&E, Investment, and Operations, and are presented in summary form as well as spread into yearly values.

Payload costs are comprised of the contributions from the spacecraft and the mission equipment. Included in the spacecraft are these subsystems:

(1) Structures
(2) Electrical power
(3) Tracking and command
(4) Stability and control
(5) Propulsion

Estimates of basic RDT&E and unit cost are provided for each subsystem. Adjustments to these values can be made because of inheritance of development from prior programs, redundancy effects on cost, and low cost design factors. Based on quantities required for the total program, unit cost values are summed to obtain the total investment cost. Refurbishment is an operations cost and is determined from the refurbishment cost factors developed by LMSC. Values used for this study range from 32 to 39 percent of the unit cost. Annual expenditures are obtained by spreading each cost element according to the following ground rules; (1) RDT&E over four years, (2) investment-unit over three years, (3) refurbishment over two years, and (4) operations over two years.

7.2 SOLAR OBSERVATORY PROGRAM COST ANALYSIS

7.2.1 Summary

The solar observatory program was used to study and evaluate the sortie mode of operation. A total program approach was taken in which the mission objectives were met by different program concepts. Each concept involved several configuration elements. In these concepts the Shuttle-sortie 7-day mission was studied both in programmed sequence with a free flyer, and as an alternative to a free flyer solar observatory. In addition, an automated payload (OSO) was used to supplement the sortie mode.

The cost analysis was conducted on a program level rather than on a single flight basis. In order to synthesize a program, two tradeoffs were made involving (1) a sortie configuration selection, and (2) a sortie and free flyer comparison. Involved in the final tradeoffs were the evaluation of eight sortie configurations - four large solar observatories (LSO) and four austere solar observatories (ASO). The LSO instrument group features a 1.5-m photoheliograph and the ASO instrument group features a 1.0-m photoheliograph. One configuration from each concept was selected for the program level approach.

Recommendations for selection of candidate concepts were based on technical and relative cost considerations. Technical considerations included relative mission effectiveness of each configuration. Mission effectiveness was evaluated first on the ability to meet mission equipment requirements, second on the flexibility to deploy the payload and conduct the mission with minimum interference from other missions being conducted by the Shuttle, and third on man-tending to make adjustments to the mission equipment. Cost considerations include relative program costs as well as other cost sensitive characteristics. Examples are Shuttle payload bay occupancy sharing potential with other payloads, potential for using Shuttle subsystems for support, and use of non-Shuttle peculiar equipment such as a general purpose lab.

The results of the cost analysis indicated the visibility into budgetary effects and therefore the advantages of employing a total program approach for making configuration selections and for synthesis of a preferred program. For instance, the results indicate the competitiveness of the sortie mode as a potential for low cost space operations. The 7-day mission has additional potential for cost reduction. The equipment on board need to be dependable for only the 7 days between opportunities for major repairs. Minor repairs can be made on-orbit. The effects on costs should be analyzed. A significant cost driver on the solar observatory program was the mission equipment requirements for resolution and pointing accuracy. With less stringent equipment requirements the savings with the sortie mode could be maximized.

The analysis showed that the step development approach, as depicted in Case II, is estimated to cost more than the normal development, without benefit of the sortie preceding the development of the free flyer. Thus the use of sortie in this case as the first step in a 2-step process is not recommended for lowest cost.

7.2.1.1 Lower Cost Configuration Selection

Payload program costs for all the sortie configurations are displayed in Table 7-1. For comparison purposes costs were estimated for payload programs with one production unit. Brief descriptive comments are provided to identify each configuration to indicate basic characteristics. The costs shown are only for the payload-peculiar items and do not include the additional costs required for the teleoperator development, investment, or operations with configuration D, and the General Purpose Lab (GPL) development or investment with configurations G and H.

Table 7-1. Cost Comparison of Sortie Configurations
(Millions of 1971 Dollars)

Configuration		Cost for One Payload			Total	Comments
		RDT&E	Invest	OPS		
LSO	A	127	66	4	197	hard-mated to Shuttle
	B	152	79	5	236	gimbaled and torqued with Shuttle
	C	167	83	5	255	tethered free flyer uses Shuttle support
	D	188	98	6	292	free flyer with teleoperator
ASO	E	88	37	2	127	non-deployed, unmanned
	F	86	38	2	126	deployed, man-tended
	G	86	36	2	124	deployed, man-tended, inside GPL
	H	87	39	3	129	deployed, man-tended, attached to GPL

The cost trend for the LSO group in going from configurations A to D reflects an increase in design complexity and mission pointing accuracy capability. In contrast, the ASO configurations are very similar in costs because the mission equipment and capabilities are similar. Selection of preferred configurations in both the LSO and ASO groups requires technical evaluation of the mission capabilities.

In the LSO group, configurations C and D are the only contenders which can provide the platform pointing accuracy required by the mission equipment. With CMG gyros in the observatory and high flexibility both configurations can be accurately controlled. In terms of mission effectiveness D is slightly better than C because it is a free flyer with complete freedom from interference; C, however, does perform adequately. In terms of costs C is lower than D because it takes advantage of the Shuttle for subsystem support through the tether. Configuration D has a further disadvantage in that there are additional costs and operating complexities associated with the teleoperator, which is assumed to be provided. Based on lower program costs (by nearly 15 percent), lower risk, and nearly equal effectiveness configuration C is selected as the lower cost candidate for the LSO group.

In the ASO group, configuration F is selected as the most effective for the same cost. It is man-tended, which is desirable for experiment and support adjustment, repair, or data recovery, has high deployment flexibility in order to ensure mission conduct with minimum interference, and occupies a smaller volume and therefore has a higher occupancy sharing potential with other payloads. Furthermore, it is not burdened with the costs and restrictions associated with sharing a general purpose lab (GPL) such as configurations G and H. The use of the GPL requires modification costs which are of the same magnitude as those encountered in developing a sortie can. The net present value (NPV) cost estimate data displayed in Table 7-3 show that the NPV rankings are the same as the total program costs for these programs.

Cost drivers in the designs are the requirements for pointing accuracy, the self-sufficiency from Shuttle support, and the performance capability of the mission equipment. Table 7-2 shows the cost impact of pointing accuracy capability among the LSO configurations. In order to increase the pointing accuracy from 5 arc sec as provided by configuration A in the Shuttle to 1 arc sec as provided by configuration C requires a 30-percent increase in the basic satellite RDT&E and investment cost. A comparison of configurations B and F shows the impact of mission equipment performance at a constant pointing accuracy capability. The austere equipment reduces basic satellite RDT&E and investment costs by over 50 percent. Another impact is the cost savings obtained by using the Shuttle for subsystem support. A comparison of configurations C and D shows the impact to be a 13-percent reduction on basic satellite RDT&E and investment costs.

7.2.1.2 Solar Observatory Program Comparisons

Four approaches to a solar observatory program were developed and compared to evaluate the sortie mode of operations. The four cases consist of combinations of sorties/LSO, sortie/ASO, free flyers, and OSO configurations compared with a dedicated free flyer. The four programs are each rated as having satisfactory content for the solar scientific community. Total program costs are shown in Table 7-3 for the four cases considered. The program life is 12 years with the first launch scheduled in 1979 and the last launch in 1990. All cases using a sortie (Cases I, II, and IV) include an automated orbiting solar observatory (OSO) satellite. The free flyer LSO (Case III) does not use the OSO. The cases are further described in section 4 of this Volume.

The differences in the cases are the schedules for each configuration. Case I used the LSO sortie C throughout the program life. Case II starts with the LSO sortie C for the first three operational years of the program, and

Table 7-2. Impact of Sortie Characteristics on Costs

Sortie Configurations		Pointing Accuracy Reqts., arc sec	Mission Equip Type (lens diameter)	Shuttle Support Provided	Basic RDT&E + Invest. Cost $M	
					Mission Equip	Satellite
LSO	A	5	Large (1.5 m)	Yes	67	165
	B	2	(1.5 m)	Yes	67	200
	C	1	(1.5 m)	Yes	67	220
	D	1	(1.5 m)	No	67	252
ASO	F	10	Austere (1.0 m)	Yes	41	102

Table 7-3. Program Direct Cost Summary - Direct Program Costs (Millions of 1971 Dollars)

SOLAR OBSERVATORY CASE	SPACE TRANSPORTATION SYSTEM PAYLOAD TOTAL	LNCH VEH DIRECT	PROGRAM DIRECT	NET PRESENT VALUE (1)
SOLAR OBSERVATORY PROGRAMS				
CASE I (SORTIE LSO, AUTOMATED OSO)				
LSO SORTIE C	437.	100.	537.	
OSO	204.	32.	236.	
SUBTOTAL	641.	132.	773.	350.
CASE II (SORTIE LSO, ON-ORB MNT LSO, OSO)				
LSO SORTIE C	294.	42.	336.	
LSO ON-ORB MNT - FREE FLYER	500.	58.	558.	
OSO	117.	11.	128.	
SUBTOTAL	911.	111.	1022.	507.
CASE III (ON-ORB MAINT LSO)				
LSO ON-ORB MNT - FREE FLYER	682.	95.	777.	
SUBTOTAL	682.	95.	777.	374.
CASE IV (AUSTERE LSO, ON-ORB MNT LSO, OSO)				
ASO SORTIE F	183.	74.	257.	
LSO ON-ORB MNT - FREE FLYER	399.	32.	431.	
OSO	159.	21.	180.	
SUBTOTAL	741.	127.	868.	395.

(1) INFINITE HORIZON AT 10% DISCOUNT

then phases into the free flyer LSO in the fourth operational year. The automated OSO is discontinued after three operational years. Case III uses the free flyer LSO throughout the program life. Case IV starts with the ASO sortie F for the first nine operational years of the program. In the eighth operational year of the program, the free flyer LSO is introduced and the automated OSO is discontinued.

Cases I and III are similar in magnitude of costs and the lowest in total program cost. Both contain the fewest number of major developments which keeps the costs lower. Case IV ranks next in cost and Case II is the highest. Annual funding levels shown in Table 7-4 indicate that Case I has lower initial costs than Case III. In terms of NPV for the total program, Case I costs $350 million against $374 million for Case III. The values are for an infinite horizon at 10-percent discount. Thus, on a relative cost basis the LSO sortie program of Case I is competitive if not slightly better than the free flyer program of Case III. The program requiring the least funding in the initial years is Case IV with the austere solar observatory program. These results on the sortie program comparisons are summarized in Table 7-5.

7.2.1.3 Low Cost Potential of Sortie Mode

The cost estimates for the sortie program used the simple design characteristics of each sortie configuration as provided by the designers. Emphasis was placed on using a valid low cost approach without resorting to low cost (big dumb payload) factors in the payload model calculations. High complexity mission equipment of the sensor type was reduced to low complexity equipment through the use of low cost optics, relaxed thermal control requirements, and low cost structural materials. Power system development costs were reduced where power was drawn from the Shuttle. Costs for the CMGs and the environmental control were reduced by considering them similar to structures.

Table 7-4. Program Direct Cost Summary - Direct Program Costs
LSO Case - Space Transportation System (Millions of 1971 Dollars)

	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	1992
SOLAR OBSERVATORY PROGRAMS																		
CASE I (SORTIE LSO, AUTOMATED OSO)																		
LSO SORTIE C	0.	21.	92.	112.	65.	34.	36.	26.	18.	17.	34.	27.	17.	13.	13.	12.	0.	0.
OSO	0.	0.	0.	24.	49.	29.	13.	17.	17.	15.	6.	16.	17.	15.	6.	12.	0.	0.
SUBTOTAL	0.	21.	92.	136.	114.	63.	49.	43.	35.	32.	40.	43.	34.	28.	19.	24.	0.	0.
CASE II (SORTIE LSO, ON-ORB MNT LSO, OSO)																		
LSO SORTIE C	0.	20.	93.	112.	65.	30.	16.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
LSO FREE FLYER	0.	0.	0.	0.	24.	117.	141.	56.	11.	14.	30.	70.	60.	10.	11.	14.	0.	0.
OSO	0.	0.	0.	24.	50.	29.	13.	12.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
SUBTOTAL	0.	20.	93.	136.	139.	176.	170.	68.	11.	14.	30.	70.	60.	10.	11.	14.	0.	0.
CASE III (ON-ORB MAINT LSO)																		
LSO FREE FLYER	0.	28.	129.	153.	69.	20.	23.	30.	69.	59.	10.	14.	30.	72.	59.	12.	0.	0.
SUBTOTAL	0.	28.	129.	153.	69.	20.	23.	30.	69.	59.	10.	14.	30.	72.	59.	12.	0.	0.
CASE IV (AUSTERE LSO, ON-ORB MNT LSO,OSO)																		
ASO SORTIE F	0.	10.	48.	54.	39.	25.	24.	17.	11.	8.	8.	6.	7.	0.	0.	0.	0.	0.
LSO FREE FLYER	0.	0.	0.	0.	0.	0.	0.	0.	29.	127.	150.	61.	17.	16.	15.	16.	0.	0.
OSO	0.	0.	0.	25.	49.	29.	13.	17.	16.	15.	6.	10.	0.	0.	0.	0.	0.	0.
SUBTOTAL	0.	10.	48.	79.	88.	54.	37.	34.	56.	150.	164.	77.	24.	16.	15.	16.	0.	0.

Table 7-5. Conclusions of Solar Observatory Program Cost Comparisons

- TOTAL PROGRAM COSTS

  Lowest Cost

  - Cases I and III - both are similar in magnitude
  - Involves least number of development programs

  Highest Cost

  - Case II
  - Includes three major development programs

- ANNUAL FUNDING CONSIDERATIONS

  Case I vs Case III

  - Case I funding level slightly lower for first three years
  - Case I has lowest net present value

  Case IV has overall lowest funding in early years

  - Delayed scheduling for free flyer
  - Funding peaks delayed to 1984-1985 period

- PRELIMINARY GUIDELINE IMPLICATIONS

  Tradeoffs of conceptual designs should be conducted on the basis of total program costs, e.g., this LSO analysis verifies competitiveness of sortie mode

  For missions with less stringent pointing requirements the sortie mode should decrease costs

  Number of major developments is a cost driver

Because of the sortie mode of operations and the 7-day on-orbit mission further cost savings should be realized owing to reduced design life requirements, testing, simple designs, etc. Examples of potential cost reductions that should be considered in later studies are discussed below.

The structure can be designed for fabrication in a simple shape. It is not necessary to use expensive materials or fabrication techniques. In the two designs selected, LSO sortie C and ASO sortie F, the electrical power system could include simple power cables that tap into the Shuttle's electrical power system. The tracking and command subsystem might be a simple data storage bank compatible with the Shuttle's tracking, command, and communication subsystem. Simple low cost tape recorders can be used to collect data. The stability and control and mission equipment are governed by the requirements of the solar observatory, but need to operate for only seven days with high reliability. In cases where the sortie is man-tended the usefulness of man should be fully exploited by reducing the amount of automated equipment.

7.2.2 Configuration Selection

7.2.2.1 Configuration Characteristics

Eight sortie configurations designed to fly the solar observatory mission for seven days were evaluated in the configuration selection. Four were LSO and four were ASO sorties. The LSO sortie configurations are designated as sortie A, B, C, and D. All LSO sortie modules carried 4,909 kg (10,825 lb) of mission equipment and were unmanned and unpressurized. The ASO sortie configurations are designated as sortie E, F, G, and H. All ASO sortie modules carried 1,619 kg (3,570 lb) of mission equipment and can be unmanned or man-tended. The characteristics of the eight sorties are shown in Table 7-6.

Table 7-6. LSO Sortie - 7-day Mission Low Earth

LSO		ASO	
Unmanned and Unpressurized		Pointing Accuracy 10 arc sec	
Resolution Requirements = 0.1 arc sec		Resolution Requirements = 0.15 arc sec	
Config	Description	Config	Description
A	Hard-mated Rigid attach (deployable) Pointing accuracy = 5 arc sec Mission equipment Weight = 4,909 kg (10,825 lb) Length = 10.8 m (34 ft)	E	Non-deployed Unmanned Mission equipment Weight = 1,619 kg (3,570 lb) Length = 2.4 m (8 ft)
B	Gimbaled (deployable) Pointing accuracy = 2 arc sec Mission equipment Weight = 4,909 kg (10,825 lb) Length = 10.8 m (34 ft)	F	Deployed Man-tended Mission equipment Weight = 1,619 kg (3,570 lb) Length = 3.05 m (10 ft)
C	Free flyer Manipulator control Pointing accuracy = 1 arc sec Mission equipment Weight = 4,909 kg (10,825 lb) Length = 11.0 m (36 ft)	G	Deployed Man-tended In GPL Mission equipment Weight = 1,619 kg (3,570 lb) Length = 11.0 m (13 ft)
D	Free flyer Telemetry control plus Teleoperator Pointing accuracy = 1 arc sec Mission equipment Weight = 4,909 kg (10,825 lb) Length = 12.2 m (40 ft)	H	Deployed Man-tended Attached to GPL Mission equipment Weight = 1,619 kg (3,570 lb) Length = 11.0 m (13 ft)

7.2.2.2 Cost Results

The eight sortie configurations that were evaluated during the configuration selection analysis were costed by the Aerospace Payload Cost Model. No low cost or redundant cost factors were used for the estimates. For the comparison cases only one RDT&E for the sortie was paid for, a buy of one complete sortie was made, and one launch was scheduled.

In the payload cost model the payload or sortie is separated into two parts - spacecraft and mission equipment. The spacecraft is made up of the following subsystems:

(1) Structure
(2) Electrical Power
(3) Tracking and Command
(4) Stability and Control
(5) Propulsion

There is no propulsion subsystem for these sorties. The mission equipment is the solar observatory and is designated either an LSO or an ASO. Table 7-7 presents the results of the analysis on LSO sortie A. The subsystems are listed by name in the first left-hand column. "Spacecraft" is indented, since it is the sum of the subsystems listed above it. "Satellite" is also indented - it is the sum of "Spacecraft" and "Mission Equipment." Also listed in the first column are:

AGE (Aerospace Ground Equipment)
Launch Support
Ground Stations
Miscellaneous
SE and TD (Systems Engineering and Technical Direction)

The "Total" is the last item listed. Under "Total" the number of designs and re-designs for the spacecraft and the mission equipment are listed. This indicates the number of RDT&E programs that were paid for. Also

Table 7-7. LSO Sortie A - Payload Program Costs (Millions of 1971 Dollars)

LSO CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	12949	12949	TYPE,	EXO	1.000	1.000	20.5	11.5	11.5	21.	11.	0.	32.
ELECTRICAL POWER	390	390	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	0	0	ALT,	LOW ORBIT	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
STABILITY,CONTROL	1000	1000	TYPE,	3-AXIS	1.000	1.000	46.4	16.7	16.7	46.	17.	0.	63.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	14339	14339					69.4	28.7	28.7	69.	29.	0.	98.
MISSION EQUIPMENT	10825	10825	COMPLXTY,	LOW	1.000	1.000	34.8	32.1	32.1	35.	32.	0.	67.
SATELLITE	25164	25164					104.1	60.8	60.8	104.	61.	0.	165.
AGE					1.000		14.9			15.	0.	0.	15.
LAUNCH SUPPORT						1.000		4.0	4.0	0.	0.	4.	4.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										7.	4.	0.	11.
TOTAL										127.	66.	4.	197.

DESIGNS AND REDESIGNS
SPACECRAFT 1.00
MISSION EQUIPMENT 1.00
SATELLITE SCHEDULE
NEW (EXPENDABLE) 1.

under the designs and re-designs is the number of satellite launches that are scheduled. The next two columns are the weights, both dry and wet. The fourth and fifth columns present other inputs required by the payload cost model. The sixth and seventh columns are reserved for Low Cost or Redundant Cost Factors. They were not used, so they were set to one.

The eighth, ninth, and tenth columns present Basic RDT&E, Average Unit, and First Unit Cost. The average unit and the first unit cost will be the same when only one spacecraft is purchased. If more are purchased, a learning curve of 95 percent is used for only the spacecraft. There is no learning assumed for the Mission Equipment.

The last four columns under Payload Program Cost Estimates are for RDT&E, Investment, Operations, and Total for the complete payload program. The direct operating cost for the launch vehicle, in this case a Shuttle, is not included. Similar payload program results for sorties B through H are presented in Tables 7-8 through 7-14.

7.2.3 Sortie and Free Flyer Program Comparisons

The traffic model for the four solar observatory approaches as presented in Table 7-15 illustrates the characteristics of the programs in a very concise manner. All four programs provide the equivalent scientific value by use of sorties, automated payloads (OSO), and free flyers. Scheduled flights survey the normal solar activity while unscheduled flights are used to cover random flights during the years of heavy solar activity. A detailed description of the costing procedure used in each case is provided in the following paragraphs. These descriptions can be followed by the use of Table 7-15 along with the symbols provided on the schedules.

7.2.3.1 Program Characteristics

Table 7-8. LSO Sortie B - Payload Program Cost (Millions of 1971 Dollars)

LSO CASE					REDUNDANT					PAYLOAD PROGRAM			
	WEIGHTS				COST FACTOR		BASIC	AVG	FIRST	COST ESTIMATE			
SUBSYSTEM	DRY	TOTAL	OTHER INPUTS		DEV	PROD	RDTE	UNIT	UNIT	RDTE	INVEST	OPS	TOTAL
STRUCTURE	18301	18301	TYPE,	EXO	1.000	1.000	21.1	13.6	13.6	21.	14.	0.	35.
ELECTRICAL POWER	440	440	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	0	0	ALT,	LOW ORBIT	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
STABILITY,CONTROL	1605	1605	TYPE,	3-AXIS	1.000	1.000	68.6	26.7	26.7	69.	27.	0.	96.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	20346	20346					92.2	40.9	40.9	92.	42.	0.	134.
MISSION EQUIPMENT	10825	10825	COMPLXTY,	LOW	1.000	1.000	34.8	32.1	32.1	35.	32.	0.	67.
SATELLITE	31171	31171					126.9	73.0	73.0	127.	74.	0.	201.
AGE					1.000		14.9			15.	0.	0.	15.
LAUNCH SUPPORT						1.000		4.6	4.6	0.	0.	5.	5.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										9.	4.	0.	13.
TOTAL										152.	79.	5.	236.

DESIGNS AND REDESIGNS
SPACECRAFT 1.00
~~MISSION EQUIPMENT~~ ~~1.00~~
SATELLITE SCHEDULE
NEW (EXPENDABLE) 1.

Table 7-9. LSO Sortie C - Payload Program Cost (Millions of 1971 Dollars)

LSO CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	10570	10570	TYPE,	EXO	1.000	1.000	20.1	10.4	10.4	20.	10.	0.	30.
ELECTRICAL POWER	490	490	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	0	0	ALT,	LOW ORBIT	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
STABILITY,CONTROL	1875	2050	TYPE,	3-AXIS	1.000	1.000	85.0	34.1	34.1	85.	34.	0.	119.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	12935	13110					107.5	45.1	45.1	107.	45.	0.	152.
MISSION EQUIPMENT	10825	10825	COMPLXTY,	LOW	1.000	1.000	34.8	32.1	32.1	35.	32.	0.	67.
SATELLITE	23760	23935					142.3	77.2	77.2	142.	77.	0.	219.
AGE					1.000		14.9			15.	0.	0.	15.
LAUNCH SUPPORT						1.000		4.9	4.9	0.	0.	5.	5.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										9.	5.	0.	14.
TOTAL										167.	83.	5.	255.

DESIGNS AND REDESIGNS	
SPACECRAFT	1.00
MISSION EQUIPMENT	1.00
SATELLITE SCHEDULE	
NEW (EXPENDABLE)	1.

Table 7-10. LSO Sortie D - Payload Program Costs (Millions of 1971 Dollars)

LSO CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	11207	11207	TYPE,	EXO	1.000	1.000	20.2	10.7	10.7	20.	11.	0.	31.
ELECTRICAL POWER	1610	1610	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	380	380	ALT,	LOW ORBIT	1.000	1.000	17.0	12.1	12.1	17.	12.	0.	29.
STABILITY,CONTROL	1950	2100	TYPE,	3-AXIS	1.000	1.000	86.8	35.0	35.0	87.	35.	0.	122.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	15147	15297					126.5	58.3	58.3	126.	59.	0.	185.
MISSION EQUIPMENT	10825	10825	COMPLXTY,	LOW	1.000	1.000	34.8	32.1	32.1	35.	32.	0.	67.
SATELLITE	25972	26122					161.3	90.3	90.3	161.	91.	0.	252.
AGE					1.000		14.9			15.	0.	0.	15.
LAUNCH SUPPORT						1.000		5.6	5.6	0.	0.	6.	6.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										11.	6.	0.	17.
TOTAL										188.	98.	6.	292.

DESIGNS AND REDESIGNS	
SPACECRAFT	1.00
MISSION EQUIPMENT	1.00
SATELLITE SCHEDULE	
NEW (EXPENDABLE)	1.

Table 7-11. ASO Sortie E - Payload Program Cost (Millions of 1971 Dollars)

ASO CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	6180	6180	TYPE,	EXO	1.000	1.000	18.8	8.1	8.1	19.	8.	0.	27.
ELECTRICAL POWER	200	200	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	60	60	ALT,	LOW ORBIT	1.000	1.000	5.8	1.6	1.6	6.	2.	0.	8.
STABILITY,CONTROL	205	285	TYPE,	3-AXIS	1.000	1.000	20.1	4.8	4.8	20.	5.	0.	25.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	6645	6725					47.2	15.2	15.2	47.	16.	0.	63.
MISSION EQUIPMENT	3570	3570	COMPLXTY,	LOW	1.000	1.000	22.8	17.9	17.9	23.	18.	0.	41.
SATELLITE	10215	10295					70.0	33.1	33.1	70.	34.	0.	104.
ASE					1.000		11.9			12.	0.	0.	12.
LAUNCH SUPPORT						1.000		2.4	2.4	0.	0.	2.	2.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										5.	2.	0.	7.
TOTAL										88.	37.	2.	127.

DESIGNS AND REDESIGNS	
SPACECRAFT	1.00
MISSION EQUIPMENT	1.00
SATELLITE SCHEDULE	
NEW (EXPENDABLE)	1.

Table 7-12. ASO Sortie F - Payload Program Cost (Millions of 1971 Dollars)

ASO CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	9625	9625	TYPE,	EXO	1.000	1.000	19.9	10.0	10.0	20.	10.	0.	30.
ELECTRICAL POWER	200	200	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	60	60	ALT,	LOW ORBIT	1.000	1.000	5.8	1.6	1.6	6.	2.	0.	8.
STABILITY,CONTROL	205	205	TYPE,	3-AXIS	1.000	1.000	17.2	3.5	3.5	17.	4.	0.	21.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	10090	10090					45.4	15.7	15.7	45.	17.	0.	62.
MISSION EQUIPMENT	3570	3570	COMPLXTY,	LOW	1.000	1.000	22.8	17.9	17.9	23.	18.	0.	41.
SATELLITE	13660	13660					68.2	33.6	33.6	68.	35.	0.	103.
AGE					1.000		11.9			12.	0.	0.	12.
LAUNCH SUPPORT						1.000		2.5	2.5	0.	0.	2.	2.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										5.	2.	0.	7.
TOTAL										86.	38.	2.	126.

DESIGNS AND REDESIGNS
SPACECRAFT 1.00
MISSION EQUIPMENT 1.00
SATELLITE SCHEDULE
NEW (EXPENDABLE) 1.

Table 7-13. ASO Sortie G - Payload Program Cost (Millions of 1971 Dollars)

ASO CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	6715	6715	TYPE,	EXO	1.000	1.000	19.0	8.4	8.4	19.	8.	0.	27.
ELECTRICAL POWER	200	200	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	60	60	ALT,	LOW ORBIT	1.000	1.000	5.8	1.6	1.6	6.	2.	0.	8.
STABILITY,CONTROL	235	240	TYPE,	3-AXIS	1.000	1.000	18.5	4.1	4.1	18.	4.	0.	22.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	7180	7215					45.8	14.7	14.7	45.	15.	0.	60.
MISSION EQUIPMENT	3570	3570	COMPLXTY,	LOW	1.000	1.000	22.8	17.9	17.9	23.	18.	0.	41.
SATELLITE	10750	10785					68.6	32.6	32.6	68.	33.	0.	101.
AGE					1.000		11.9			12.	0.	0.	12.
LAUNCH SUPPORT						1.000		2.4	2.4	0.	0.	2.	2.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										5.	2.	0.	7.
TOTAL										86.	36.	2.	124.

DESIGNS AND REDESIGNS
SPACECRAFT 1.00
MISSION EQUIPMENT 1.00
SATELLITE SCHEDULE
NEW (EXPENDABLE) 1.

Table 7-14. ASO Sortie H - Payload Program Cost (Millions of 1971 Dollars)

ASO CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	11870	11870	TYPE,	EXO	1.000	1.000	20.3	11.0	11.0	20.	11.	0.	31.
ELECTRICAL POWER	200	200	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	60	60	ALT,	LOW ORBIT	1.000	1.000	5.8	1.6	1.6	6.	2.	0.	8.
STABILITY,CONTROL	205	240	TYPE,	3-AXIS	1.000	1.000	18.5	4.1	4.1	18.	4.	0.	22.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	12335	12370					47.1	17.3	17.3	46.	18.	0.	64.
MISSION EQUIPMENT	3570	3570	COMPLXTY,	LOW	1.000	1.000	22.8	17.9	17.9	23.	18.	0.	41.
SATELLITE	15905	15940					69.9	35.2	35.2	69.	36.	0.	105.
AGE					1.000		11.9			12.	0.	0.	12.
LAUNCH SUPPORT						1.000		2.6	2.6	0.	0.	3.	3.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										5.	2.	0.	7.
TOTAL										87.	39.	3.	129.

DESIGNS AND REDESIGNS	
SPACECRAFT	1.00
MISSION EQUIPMENT	1.00
SATELLITE SCHEDULE	
NEW (EXPENDABLE)	1.

Table 7-15. Traffic Models for NASA Solar Observatory Program

No.	Program Approach	Type Payload	Schedule, Year 79	80	81	82	83	84	85	86	87	88	89	90
I	Sortie	scheduled sortie (E)	2	2	2	1	1	1	1	1	1	1	1	2
		unscheduled sortie	1	1					1					
		automated P/L (OSO)*		1		1		1		1		1		1
II	Sortie - Free Flyer (same exp pkg)	scheduled sortie (E)	2	2	2									
		unscheduled sortie	1	1										
		free flyer (LSO)				1								
		manned visits					1	1	1	1	1	1	1	2
		automated P/L (OSO)		1		1								
III	Free Flyer	free flyer (LSO)	1											
		manned visits	1	2	2	1	1	2	1	1	1	1	2	2
IV	Sortie + Free Flyer (different exp pkg)	scheduled sortie (F)	2	2	2	1	1	1	1	1	1			
		unscheduled sortie	1	1										
		free flyer (LSO)								1				
		manned visits									1	1	1	2
		automated P/L (OSO)		1		1		1		1				

*orbiting solar observatory program in operation since 1962

payload R & D, mission equipment R&D

Case I

In Case I the LSO sortie C is used throughout the entire program. Program charges are determined in the following manner. Complete payload (spacecraft and mission equipment) R&D and unit costs for the LSO sortie C are charged initially to the program. The R&D cost is spread forward over four years starting in 1976 and the unit cost is spread forward over three years starting in 1977. Two more mission equipment R&Ds scheduled for 1983 and 1987 are charged to the program and spread over four years starting in 1980 and 1984. Seven-day missions are flown twice in each of the years 1979, 1980, 1981, and 1990. One seven-day mission is flown in each of the years 1982 through 1989. An unscheduled seven-day mission is flown in each of the years of 1979, 1980, and 1985. The operations costs are spread over two years starting in the year before each launch.

An automated OSO is maintained on orbit from 1980 to 1990 by periodic launches and refurbishments. A complete payload R&D and unit cost for the OSO are charged initially to the program and spread over three years starting in 1978. A new OSO is placed on order in 1982 and the investment cost is spread over three years starting in 1980. One mission equipment R&D is charged to the program in each of the years 1984 and 1988. The OSO is refurbished in each of the years 1986 and 1990.

The results of the payload program cost for the LSO sortie C and automated OSO are shown in Tables 7-16 and 7-17. Costs associated with the launch schedules are provided in Tables 7-18 and 7-19 and summed to obtain the program costs.

Case II

In Case II the LSO sortie C goes into operation in 1979 and is supported by an automated OSO launched in 1980. The free flyer becomes operational in

Table 7-16. LSO Sortie C - Payload Program Cost (Millions of 1971 Dollars)

SOLAR OBSERVATORY CASE I

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	10570	10570	TYPE,	EXO	1.000	1.000	20.1	10.4	10.4	20.	10.	0.	30.
ELECTRICAL POWER	490	490	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	0	0	ALT,	LOW ORBIT	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
STABILITY,CONTROL	1875	2050	TYPE,	3-AXIS	1.000	1.000	85.0	34.1	34.1	85.	34.	0.	119.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	12935	13110					107.5	45.1	45.1	107.	45.	0.	152.
MISSION EQUIPMENT	10825	10825	COMPLXTY,	LOW	1.000	1.000	34.8	32.1	32.1	104.	32.	0.	136.
SATELLITE	23760	23935					142.3	77.2	77.2	211.	77.	108.	396.
AGE					1.000		14.9			15.	0.	0.	15.
LAUNCH SUPPORT						1.000		4.9	4.9	0.	0.	5.	5.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										14.	5.	0.	19.
TOTAL										241.	83.	113.	437.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00				1.00				1.00					3.0
SATELLITE SCHEDULE														
NEW (EXPENDABLE)	1.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	1.
MAINTENANCE FLTS	.150	.200	.150	.100	.100	.100	.100	.100	.100	.100	.100	.1000	.0000	1.400

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	21.	71.	58.	17.	4.	16.	13.	4.	4.	16.	13.	4.	0.	0.	0.	0.	241.
INVESTMENT	0.	0.	21.	46.	16.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	83.
OPERATIONS	0.	0.	0.	0.	16.	14.	10.	8.	8.	8.	8.	8.	8.	8.	8.	1.	0.	113.
TOTAL	0.	21.	92.	112.	49.	18.	26.	21.	12.	12.	24.	21.	12.	8.	8.	1.	0.	437.

Table 7-17. OSO - Payload Program Cost (Millions of 1971 Dollars)

SOLAR OBSERVATORY CASE I

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		LOW COST COST FACTOR DEV	LOW COST COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	550	550	TYPE,	EXO	.560	.590	8.9	2.0	2.0	9.	4.	0.	13.
ELECTRICAL POWER	350	350	WATTS,	300.	.650	.790	4.3	.9	.9	4.	2.	0.	6.
TRACKING,COMMAND	160	160	ALT,	LOW ORBIT	.700	.850	10.0	4.2	4.2	10.	8.	0.	18.
STABILITY,CONTROL	270	420	TYPE,	3-AXIS	.610	.830	19.6	5.9	5.9	20.	12.	0.	32.
PROPULSION	0	0	TOT.IMP.	0.	.850	.750	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1330	1480					42.8	12.9	12.9	43.	26.	0.	69.
MISSION EQUIPMENT	500	500	COMPLXTY,	MED	.630	.830	14.7	6.8	6.8	44.	22.	0.	66.
SATELLITE	1830	1980					57.5	19.7	19.7	87.	48.	31.	166.
AGE					.710		4.4			4.	0.	0.	4.
LAUNCH SUPPORT						.740		4.0	4.0	0.	0.	24.	24.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										6.	1.	1.	8.
TOTAL										98.	50.	56.	204.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT		1.00												1.0
MISSION EQUIPMENT		1.00				1.00				1.00				3.0
SATELLITE SCHEDULE														
NEW (LO COST REUSE)	0.	1.	0.	1.	0.	0.	0.	0.	0.	0.	0.	0.	0.	2.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	1.	0.	1.	0.	1.	0.	1.	0.	4.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	0.	0.	17.	36.	13.	0.	4.	9.	3.	0.	4.	9.	3.	0.	0.	0.	98.
INVESTMENT	0.	0.	0.	7.	11.	9.	11.	5.	2.	1.	0.	1.	2.	1.	0.	0.	0.	50.
OPERATIONS	0.	0.	0.	0.	2.	2.	2.	2.	6.	6.	6.	6.	6.	6.	6.	6.	0.	56.
TOTAL	0.	0.	0.	24.	49.	24.	13.	11.	17.	10.	6.	11.	17.	10.	6.	6.	0.	204.

Table 7-18. Solar Observatory Case I - LSO Sortie C

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
LSO SORTIE OSSA
LAUNCHED FROM ETR

LSO CASE
SPACE TRANSPORTATION SYSTEM
SCHEDULED QUANTITIES — PROGRAM DIRECT COST
LAUNCH VEHICLES

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB					SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	21.	0.	21.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	92.	0.	92.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	112.	0.	112.
1979	1.	0.	0.0	0.0	0.0	0.0	1.5	0.0	0.0	49.	16.	65.
1980	0.	0.	0.0	0.0	0.0	0.0	1.5	0.0	0.0	18.	16.	34.
1981	0.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	26.	10.	36.
1982	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	21.	5.	26.
1983	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	12.	6.	18.
1984	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	12.	5.	17.
1985	0.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	24.	10.	34.
1986	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	21.	6.	27.
1987	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	12.	5.	17.
1988	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	8.	5.	13.
1989	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	8.	5.	13.
1990	0.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	1.	11.	12.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	1.	0.	0.0	0.0	0.0	0.0	9.5	0.0	0.0	437.	100.	537.

Table 7-19. Solar Observatory Case I - OSO

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
OSO OSS

LAUNCHED FROM ETR

LSO CASE
SPACE TRANSPORTATION SYSTEM
SCHEDULED QUANTITIES

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	LAUNCH VEHICLES				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	24.	0.	24.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	49.	0.	49.
1980	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	24.	5.	29.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	13.	0.	13.
1982	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	11.	6.	17.
1983	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	17.	0.	17.
1984	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	10.	5.	15.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	0.	6.
1986	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	11.	5.	16.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	17.	0.	17.
1988	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	10.	5.	15.
1989	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	0.	6.
1990	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	6.	6.	12.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	2.	4.	0.0	0.0	0.0	0.0	3.0	0.0	0.0	204.	32.	236.

1982 and replaces the LSO sortie C. A complete payload R&D and unit costs for the LSO sortie C are charged initially to the program. The R&D costs are spread over four years starting in 1976 and the unit cost is spread over three years starting in 1977. No more new developments are charged to the program after the initial buy since the LSO sortie C goes out of service in 1981. Seven-day missions are flown twice in each of the years 1979, 1980, and 1981 and one unscheduled seven-day mission is flown by the LSO sortie C in each of the years 1979 and 1980. These operations costs are spread over two years starting in the year before the launch.

A complete payload R&D and unit costs for the automated OSO are charged initially to the program with the R&D and unit costs, spread over three years starting in 1978. In 1982 a new OSO is placed in orbit and its cost is spread over three years starting in 1980. One automated OSO is launched in 1980 and 1982. The cost is spread over two years starting in the year before the launch.

The free flyer goes into operation in 1982. The spacecraft R&D unit costs are charged to the program, but only 25 percent of the free flyer's mission equipment R&D and a complete free flyer's mission equipment unit costs are charged to the program since it obtains technology development data from the LSO sortie. The R&D costs are spread over four years starting in 1979. The unit costs are spread over three years starting in 1980. In 1987 the free flyer's mission equipment is redesigned and the spacecraft is refurbished. The free flyer is visited once each year from 1983 through 1990, with the exceptions of 1987 and 1990 when it is visited twice yearly.

The results of the payload program cost for the LSO sortie C, the free flyer, and the automated OSO are shown in Tables 7-20 through 7-22. The program direct costs for Case II are shown in Tables 7-23 through 7-25.

Table 7-20. LSO Sortie C - Payload Program Cost (Millions of 1970 Dollars)

SOLAR OBSERVATORY CASE II

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	10570	10570	TYPE,	EXO	1.000	1.000	20.1	10.4	10.4	20.	10.	0.	30.
ELECTRICAL POWER	490	490	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	0	0	ALT,	LOW ORBIT	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
STABILITY,CONTROL	1875	2050	TYPE,	3-AXIS	1.000	1.000	85.0	34.1	34.1	85.	34.	0.	119.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	12935	13110					107.5	45.1	45.1	107.	45.	0.	152.
MISSION EQUIPMENT	10825	10825	COMPLXTY,	LOW	1.000	1.000	34.8	32.1	32.1	35.	32.	0.	67.
SATELLITE	23760	23935					142.3	77.2	77.2	142.	77.	39.	258.
AGE					1.000		14.9			15.	0.	0.	15.
LAUNCH SUPPORT						1.000		4.9	4.9	0.	0.	5.	5.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										9.	5.	0.	14.
TOTAL										167.	83.	44.	294.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00													1.0
SATELLITE SCHEDULE														
NEW (EXPENDABLE)	1.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	1.
MAINTENANCE FLTS	.150	.200	.150	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	.500

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	20.	72.	58.	17.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	167.
INVESTMENT	0.	0.	21.	46.	16.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	83.
OPERATIONS	0.	0.	0.	8.	16.	14.	6.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	44.
TOTAL	0.	20.	93.	112.	49.	14.	6.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	294.

Table 7-21. LSO Free Flyer - Payload Program Costs (Millions of 1971 Dollars)

SOLAR OBSERVATORY CASE II

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	9340	9340	TYPE,	EXO	1.000	1.000	19.8	9.8	9.8	20.	10.	0.	30.
ELECTRICAL POWER	2850	2850	WATTS,	1500.	1.000	1.000	9.8	2.2	2.2	10.	2.	0.	12.
TRACKING,COMMAND	1100	1100	ALT,	LOW ORBIT	1.000	1.000	42.2	35.5	35.5	42.	35.	0.	77.
STABILITY,CONTROL	1865	2065	TYPE,	3-AXIS	1.000	1.000	85.5	34.4	34.4	86.	34.	0.	120.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	15155	15355					157.4	81.9	81.9	158.	81.	0.	239.
MISSION EQUIPMENT	9934	9934	COMPLXTY,	LOW	1.000	1.000	33.6	30.7	30.7	42.	52.	0.	94.
SATELLITE	25089	25289					191.0	112.6	112.6	200.	133.	108.	441.
AGE					1.000		21.5			22.	0.	0.	22.
LAUNCH SUPPORT						1.000		6.6	6.6	0.	0.	13.	13.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	2.	0.	3.
SE AND TD										13.	7.	1.	21.
TOTAL										236.	142.	122.	500.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT				1.00										1.0
MISSION EQUIPMENT				.25					1.00					1.3
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	0.	0.	0.	1.	0.	0.	0.	0.	0.	0.	0.	0.	0.	1.
REFURB (RATE=.320)	0.	0.	0.	0.	0.	0.	0.	0.	1.	0.	0.	0.	0.	1.
MAINTENANCE FLTS	0.000	0.000	0.000	0.000	.100	.050	.100	.050	.100	.050	.100	.090	0.000	.640

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	0.	0.	0.	24.	86.	70.	20.	0.	4.	15.	13.	4.	0.	0.	0.	0.	236.
INVESTMENT	0.	0.	0.	0.	0.	31.	67.	24.	0.	0.	5.	11.	4.	0.	0.	0.	0.	142.
OPERATIONS	0.	0.	0.	0.	0.	0.	4.	7.	5.	5.	5.	41.	41.	5.	6.	3.	0.	122.
TOTAL	0.	0.	0.	0.	24.	117.	141.	51.	5.	9.	25.	65.	49.	5.	6.	3.	0.	500.

Table 7-22. OSO - Payload Program Cost (Millions of 1971 Dollars)

SOLAR OBSERVATORY CASE II

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		LOW COST FACTOR DEV	LOW COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	550	550	TYPE,	EXO	.560	.590	8.9	2.0	2.0	9.	4.	0.	13.
ELECTRICAL POWER	350	350	WATTS,	300.	.650	.790	4.3	.9	.9	4.	2.	0.	6.
TRACKING,COMMAND	160	160	ALT,	LOW ORBIT	.700	.850	10.0	4.2	4.2	10.	8.	0.	18.
STABILITY,CONTROL	270	420	TYPE,	3-AXIS	.610	.830	19.6	5.9	5.9	20.	12.	0.	32.
PROPULSION	0	0	TOT.IMP.	0.	.850	.750	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1330	1480					42.8	12.9	12.9	43.	26.	0.	69.
MISSION EQUIPMENT	500	500	COMPLXTY,	MED	.630	.830	14.7	6.8	6.8	15.	14.	0.	29.
SATELLITE	1830	1980					57.5	19.7	19.7	58.	40.	0.	98.
AGE					.710		4.4			4.	0.	0.	4.
LAUNCH SUPPORT						.740		4.0	4.0	0.	0.	8.	8.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										4.	1.	0.	5.
TOTAL										67.	42.	8.	117.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT		1.00												1.0
MISSION EQUIPMENT		1.00												1.0
SATELLITE SCHEDULE														
NEW (LO COST EXPND)	0.	1.	0.	1.	0.	0.	0.	0.	0.	0.	0.	0.	0.	2.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	0.	0.	17.	37.	13.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	67.
INVESTMENT	0.	0.	0.	7.	11.	9.	11.	4.	0.	0.	0.	0.	0.	0.	0.	0.	0.	42.
OPERATIONS	0.	0.	0.	0.	2.	2.	2.	2.	0.	0.	0.	0.	0.	0.	0.	0.	0.	8.
TOTAL	0.	0.	0.	24.	50.	24.	13.	6.	0.	0.	0.	0.	0.	0.	0.	0.	0.	117.

Table 7-23. Solar Observatory Case II - LSO Sortie C

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
LSO SORTIE OSS
LAUNCHED FROM ETR

LSO CASE
SPACE TRANSPORTATION SYSTEM
SCHEDULED QUANTITIES / PROGRAM DIRECT COST
LAUNCH VEHICLES

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB					SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	20.	0.	20.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	93.	0.	93.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	112.	0.	112.
1979	1.	0.	0.0	0.0	0.0	0.0	1.5	0.0	0.0	49.	16.	65.
1980	0.	0.	0.0	0.0	0.0	0.0	1.5	0.0	0.0	14.	16.	30.
1981	0.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	6.	10.	16.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1983	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1989	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	1.	0.	0.0	0.0	0.0	0.0	4.0	0.0	0.0	294.	42.	336.

Table 7-24. Solar Observatory Case II - Free Flyer

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
LSO ON-ORB MNT OSS
LAUNCHED FROM ETR

LSO CASE
SPACE TRANSPORTATION SYSTEM

	SCHEDULED QUANTITIES										PROGRAM DIRECT COST		
			LAUNCH VEHICLES										
FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB					SHTL SHTL	TUG	TUG EXP		PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		24.	0.	24.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		117.	0.	117.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		141.	0.	141.
1982	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0		51.	5.	56.
1983	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0		5.	6.	11.
1984	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0		9.	5.	14.
1985	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0		25.	5.	30.
1986	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0		65.	5.	70.
1987	0.	1.	0.0	0.0	0.0	0.0	1.0	0.0	0.0		49.	11.	60.
1988	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0		5.	5.	10.
1989	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0		6.	5.	11.
1990	0.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0		3.	11.	14.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
TOTAL	1.	1.	0.0	0.0	0.0	0.0	5.5	0.0	0.0		500.	58.	558.

Table 7-25. Solar Observatory Case II - OSO

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
OSO OSS
LAUNCHED FROM ETR

LSO CASE
SPACE TRANSPORTATION SYSTEM

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	LAUNCH VEHICLES				SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
	SCHEDULED QUANTITIES									PROGRAM DIRECT COST		
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	24.	0.	24.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	50.	0.	50.
1980	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	24.	5.	29.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	13.	0.	13.
1982	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	6.	6.	12.
1983	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1989	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	2.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	117.	11.	128.

Case III

In Case III only the free flyer is used throughout the entire program. A complete payload R&D and unit costs for the free flyer are charged to the program with the R&D cost spread over four years starting in 1976 and the unit cost spread over three years starting in 1977. Two mission equipments R&Ds are charged to the program and spread over four years starting in 1981 and 1986. The free flyer is visited each year from 1979 through 1990 and twice in each of the years 1980, 1981, and 1990. In each of the years 1984 and 1989 the free flyer receives a revisit and a refurbishment flight. The launch operations costs are spread over two years starting in the year before the launch. The refurbishment cost is spread over three years starting in the years 1982 and 1987.

The results of the LSO free flyer are shown in Table 7-26 and the program direct cost for Case III is shown in Table 7-27.

Case IV

In Case IV the ASO sortie F goes into operation in 1979 and is supported by the automated OSO that goes operational in 1980. The free flyer goes operational in 1986, a year prior to termination of the sortie flights.

A complete R&D and unit costs for the ASO sortie F are charged initially to the program, with the R&D costs spread over four years starting in 1976 and the unit cost spread over three years starting in 1977. In 1983 a mission equipment R&D is charged to the program and spread forward over four years starting in 1980. A refurbishment is charged to the program and spread over three years starting in 1984. Seven-day missions are flown twice in each of the years 1979, 1980, and 1981 and one seven-day mission is flown in each of the years 1982 through 1987. Also, one unscheduled seven-day mission is flown by the ASO sortie F in each of the years 1979

Table 7-26. Free Flyer - Payload Program Cost (Millions of 1971 Dollars)

SOLAR OBSERVATORY CASE III

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	9340	9340	TYPE,	EXO	1.000	1.000	19.8	9.8	9.8	20.	10.	0.	30.
ELECTRICAL POWER	2850	2850	WATTS,	1500.	1.000	1.000	9.8	2.2	2.2	10.	2.	0.	12.
TRACKING,COMMAND	1100	1100	ALT,	LOW ORBIT	1.000	1.000	42.2	35.5	35.5	42.	35.	0.	77.
STABILITY,CONTROL	1865	2065	TYPE,	3-AXIS	1.000	1.000	85.5	34.4	34.4	86.	34.	0.	120.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	15155	15355					157.4	81.9	81.9	158.	81.	0.	239.
MISSION EQUIPMENT	9934	9934	COMPLXTY,	LOW	1.000	1.000	33.6	30.7	30.7	101.	73.	0.	174.
SATELLITE	25089	25289					191.0	112.6	112.6	259.	154.	199.	612.
AGE					1.000		21.5			22.	0.	0.	22.
LAUNCH SUPPORT						1.000		6.6	6.6	0.	0.	20.	20.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	2.	0.	3.
SE AND TD										17.	7.	1.	25.
TOTAL										299.	163.	220.	682.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00					1.00					1.00			3.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	1.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	1.
REFURB (RATE=.320)	0.	0.	0.	0.	0.	1.	0.	0.	0.	0.	1.	0.	0.	2.
MAINTENANCE FLTS	.100	.150	.150	.100	.050	.100	.050	.100	.050	.100	.100	.080	0.000	1.130

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	28.	97.	79.	23.	0.	4.	15.	13.	4.	0.	4.	15.	13.	4.	0.	0.	299.
INVESTMENT	0.	0.	32.	67.	24.	0.	0.	5.	11.	4.	0.	0.	5.	11.	4.	0.	0.	163.
OPERATIONS	0.	0.	0.	7.	11.	10.	8.	5.	40.	40.	5.	5.	5.	42.	41.	1.	0.	220.
TOTAL	0.	28.	129.	153.	58.	10.	12.	25.	64.	48.	5.	9.	25.	66.	49.	1.	0.	682.

Table 7-27. Solar Observatory Case III, Free Flyer

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
LSO ON-ORB MNT OSS
LAUNCHED FROM ETR

LSO CASE
SPACE TRANSPORTATION SYSTEM
SCHEDULED QUANTITIES / PROGRAM DIRECT COST

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	LAUNCH VEHICLES				SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	28.	0.	28.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	129.	0.	129.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	153.	0.	153.
1979	1.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	58.	11.	69.
1980	0.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	10.	10.	20.
1981	0.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	12.	11.	23.
1982	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	25.	5.	30.
1983	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	64.	5.	69.
1984	0.	1.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	48.	11.	59.
1985	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	5.	5.	10.
1986	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	9.	5.	14.
1987	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	25.	5.	30.
1988	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	66.	6.	72.
1989	0.	1.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	49.	10.	59.
1990	0.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	1.	11.	12.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	1.	2.	0.0	0.0	0.0	0.0	9.0	0.0	0.0	682.	95.	777.

and 1980. These operations costs are spread over two years starting in the year before the launch.

A complete payload R&D and unit costs for the automated OSO are charged to the program, with the R&D and unit costs spread over three years starting in 1978. A new mission equipment R&D for the automated OSO is charged to the program and spread over three years starting in 1982. A refurbishment cost is charged to the program and spread over two years starting in the years 1983 and 1985.

The LSO free flyer goes into operation in 1986. A complete spacecraft and mission equipment R&D and unit costs for the LSO free flyer are charged to the program. The LSO free flyer does not inherit technology development data from the ASO sortie. The R&D costs are spread over four years starting in 1983. The unit costs are spread over three years starting in 1984. One manned maintenance visit is flown in each of the years 1987, 1988, and 1989; two are flown in 1990. These operations costs are spread over two years starting in the year before the launch.

The results of the payload program cost for the ASO sortie F, the free flyer, and the automated OSO are shown in Tables 7-28 through 7-30. The program direct cost for Case IV is shown in Tables 7-31 through 7-33.

7.2.3.2 Total Program Cost

Case I and Case III are the lowest in total cost. They are similar in magnitude with Case I slightly lower in NPV. The LSO sortie C and the automated OSO are used in Case I. The free flyer is used in Case III. Each of these cases involves the least number of development programs.

Case II,which has three major developments, has the highest cost. It uses the LSO sortie C, the automated OSO, and a free flyer. The free flyer is phased in shortly after the sortie, and gains from its technology.

Table 7-28. ASO Sortie F - Payload Program Costs
(Millions of 1971 Dollars)

SOLAR OBSERVATORY CASE IV

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	9625	9625	TYPE,	EXO	1.000	1.000	19.9	10.0	10.0	20.	10.	0.	30.
ELECTRICAL POWER	200	200	WATTS,	1500.	1.000	1.000	2.4	.6	.6	2.	1.	0.	3.
TRACKING,COMMAND	60	60	ALT,	LOW ORBIT	1.000	1.000	5.8	1.6	1.6	6.	2.	0.	8.
STABILITY,CONTROL	205	205	TYPE,	3-AXIS	1.000	1.000	17.2	3.5	3.5	17.	4.	0.	21.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	10090	10090					45.4	15.7	15.7	45.	17.	0.	62.
MISSION EQUIPMENT	3570	3570	COMPLXTY,	LOW	1.000	1.000	22.8	17.9	17.9	46.	18.	0.	64.
SATELLITE	13660	13660					68.2	33.6	33.6	91.	35.	33.	159.
AGE					1.000		11.9			12.	0.	0.	12.
LAUNCH SUPPORT						1.000		2.5	2.5	0.	0.	2.	2.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										6.	2.	0.	8.
TOTAL										110.	38.	35.	183.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00				1.00									2.0
SATELLITE SCHEDULE														
NEW (EXPENDABLE)	1.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	1.
MAINTENANCE FLTS	.150	.200	.150	.100	.100	.100	.100	.050	.030	0.000	0.000	0.000	0.000	.980

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	10.	37.	30.	9.	3.	10.	9.	2.	0.	0.	0.	0.	0.	0.	0.	0.	110.
INVESTMENT	0.	0.	11.	20.	7.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	38.
OPERATIONS	0.	0.	0.	4.	7.	6.	4.	3.	3.	3.	3.	1.	1.	0.	0.	0.	0.	35.
TOTAL	0.	10.	48.	54.	23.	9.	14.	12.	5.	3.	3.	1.	1.	0.	0.	0.	0.	183.

Table 7-29. Free Flyer - Payload Program Costs
(Millions of 1971 Dollars)

SOLAR OBSERVATORY CASE IV

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	9340	9340	TYPE,	EXO	1.000	1.000	19.8	9.8	9.8	20.	10.	0.	30.
ELECTRICAL POWER	2850	2850	WATTS,	1500.	1.000	1.000	9.8	2.2	2.2	10.	2.	0.	12.
TRACKING,COMMAND	1100	1100	ALT,	LOW ORBIT	1.000	1.000	42.2	35.5	35.5	42.	35.	0.	77.
STABILITY,CONTROL	1865	2065	TYPE,	3-AXIS	1.000	1.000	85.5	34.4	34.4	86.	34.	0.	120.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	15155	15355					157.4	81.9	81.9	158.	81.	0.	239.
MISSION EQUIPMENT	9934	9934	COMPLXTY,	LOW	1.000	1.000	33.6	30.7	30.7	34.	31.	0.	65.
SATELLITE	25089	25289					191.0	112.6	112.6	192.	112.	43.	347.
AGE					1.000		21.5			22.	0.	0.	22.
LAUNCH SUPPORT						1.000		6.6	6.6	0.	0.	7.	7.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	2.	0.	3.
SE AND TD										13.	7.	0.	20.
TOTAL										228.	121.	50.	399.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT								1.00						1.0
MISSION EQUIPMENT								1.00						1.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	0.	0.	0.	0.	0.	0.	0.	1.	0.	0.	0.	0.	0.	1.
MAINTENANCE FLTS	0.0000	.0000	.0000	.0000	.0000	.0000	.0000	.000	.100	.100	.100	.0800	.0000	.380

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	0.	0.	0.	0.	0.	0.	0.	29.	97.	79.	23.	0.	0.	0.	0.	0.	228.
INVESTMENT	0.	0.	0.	0.	0.	0.	0.	0.	0.	30.	67.	24.	0.	0.	0.	0.	0.	121.
OPERATIONS	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	4.	9.	11.	11.	10.	5.	0.	50.
TOTAL	0.	0.	0.	0.	0.	0.	0.	0.	29.	127.	150.	56.	11.	11.	10.	5.	0.	399.

Table 7-30. OSO - Payload Program Cost
(Millions of 1971 Dollars)

SOLAR OBSERVATORY CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		LOW COST COST FACTOR DEV	LOW COST COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	550	550	TYPE,	EXO	.560	.590	8.9	2.0	2.0	9.	4.	0.	13.
ELECTRICAL POWER	350	350	WATTS,	300.	.650	.790	4.3	.9	.9	4.	2.	0.	6.
TRACKING,COMMAND	160	160	ALT,	LOW ORBIT	.700	.850	10.0	4.2	4.2	10.	8.	0.	18.
STABILITY,CONTROL	270	420	TYPE,	3-AXIS	.610	.830	19.6	5.9	5.9	20.	12.	0.	32.
PROPULSION	0	0	TOT.IMP.	0.	.850	.750	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1330	1480					42.8	12.9	12.9	43.	26.	0.	69.
MISSION EQUIPMENT	500	500	COMPLXTY,	MED	.630	.830	14.7	6.8	6.8	29.	18.	0.	47.
SATELLITE	1830	1980					57.5	19.7	19.7	72.	44.	15.	131.
AGE					.710		4.4			4.	0.	0.	4.
LAUNCH SUPPORT						.740		4.0	4.0	0.	0.	16.	16.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										5.	1.	0.	6.
TOTAL										82.	46.	31.	159.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT		1.00												1.0
MISSION EQUIPMENT		1.00				1.00								2.0
SATELLITE SCHEDULE														
NEW (LO COST REUSE)	0.	1.	0.	1.	0.	0.	0.	0.	0.	0.	0.	0.	0.	2.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	1.	0.	1.	0.	0.	0.	0.	0.	2.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	0.	0.	18.	36.	13.	0.	4.	8.	3.	0.	0.	0.	0.	0.	0.	0.	82.
INVESTMENT	0.	0.	0.	7.	11.	9.	11.	5.	2.	1.	0.	0.	0.	0.	0.	0.	0.	46.
OPERATIONS	0.	0.	0.	0.	2.	2.	2.	2.	6.	6.	6.	5.	0.	0.	0.	0.	0.	31.
TOTAL	0.	0.	0.	25.	49.	24.	13.	11.	16.	10.	6.	5.	0.	0.	0.	0.	0.	159.

Table 7-31. Solar Observatory Case IV, ASO Sortie F

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
LSO AUSTERE OSS
LAUNCHED FROM ETR

LSO CASE
SPACE TRANSPORTATION SYSTEM
SCHEDULED QUANTITIES
LAUNCH VEHICLES

PROGRAM DIRECT COST

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB					SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	10.	0.	10.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	48.	0.	48.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	54.	0.	54.
1979	1.	0.	0.0	0.0	0.0	0.0	1.5	0.0	0.0	23.	16.	39.
1980	0.	0.	0.0	0.0	0.0	0.0	1.5	0.0	0.0	9.	16.	25.
1981	0.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	14.	10.	24.
1982	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	12.	5.	17.
1983	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	5.	6.	11.
1984	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	3.	5.	8.
1985	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	3.	5.	8.
1986	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	1.	5.	6.
1987	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	1.	6.	7.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1989	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	1.	0.	0.0	0.0	0.0	0.0	7.0	0.0	0.0	183.	74.	257.

Table 7-32. Solar Observatory Case IV, Free Flyer

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
LSO ON-ORB MNT OSS
LAUNCHED FROM ETR

LSO CASE
SPACE TRANSPORTATION SYSTEM

FISCAL YEAR	SCHEDULED QUANTITIES PAYLOADS NEW	REFURB	LAUNCH VEHICLES				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1983	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	29.	0.	29.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	127.	0.	127.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	150.	0.	150.
1986	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	56.	5.	61.
1987	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	11.	6.	17.
1988	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	11.	5.	16.
1989	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	10.	5.	15.
1990	0.	0.	0.0	0.0	0.0	0.0	1.0	0.0	0.0	5.	11.	16.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	1.	0.	0.0	0.0	0.0	0.0	3.0	0.0	0.0	399.	32.	431.

Table 7-33. Solar Observatory Case IV, OSO

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
OSO OSS
LAUNCHED FROM ETR

LSO CASE
SPACE TRANSPORTATION SYSTEM

FISCAL YEAR	SCHEDULED QUANTITIES PAYLOADS NEW	REFURB	LAUNCH VEHICLES				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	25.	0.	25.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	49.	0.	49.
1980	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	24.	5.	29.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	13.	0.	13.
1982	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	11.	6.	17.
1983	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	16.	0.	16.
1984	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	10.	5.	15.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	0.	6.
1986	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	5.	5.	10.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1989	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	2.	2.	0.0	0.0	0.0	0.0	2.0	0.0	0.0	159.	21.	180.

Case IV falls between the highest and the lowest. It uses the ASO sortie F, the automated OSO, and the free flyer. The free flyer is phased in four years later than the free flyer in Case II and does not benefit from the development of the ASO sortie F. The difference between Case IV and Case II is explained as follows.

In Case IV the ASO sortie's development and investment is less than the development and investment of the LSO sortie used in Case II. The R&D and investment for the free flyer used in both cases are about the same, but the operations cost for the free flyer is less for Case IV. The R&D for the automated OSO is more for Case IV than Case II because of extra mission equipment. The automated OSO is in operation longer in Case IV than Case II, so the operations cost is more. The overall effect is that longer use of lower cost configurations such as ASO will provide lower total program costs.

7.3 SYSTEM DEMONSTRATION PROGRAM COST ANALYSIS

7.3.1 Summary

A cost analysis was conducted on two system demonstration programs. The first program examined the effect of traffic demand on a Tracking and Data Relay Satellite (TDRS) program. Tradeoffs for this program involved a cost comparison of expendable or reusable operational modes, configuration designs, and system approaches to meet increasing demand. The second program compared the cost of conducting a System Test Satellite program with expendable or reusable configurations.

The 4.6-m (15-ft) diameter satellite (configuration D) configured for multiple launches on the Shuttle/upper stage and applied to the TDRS demonstration programs is estimated to have the lowest relative life cycle cost for each case considered. The cost driver is the reduction in launch charges due to multiple launches, compared to the next best configuration (configuration C).

In the TDRS program there are sufficient refurbishments to make reuse payoffs, whereas in the System Test Satellite program the refurbishments are too few to generate savings for the reusable mode for NASA. If the non-NASA users of the System Test Satellites accepted the demonstration and took over the program, the user would probably benefit from a reusable satellite approach. This possibility was not studied. These comparisons indicate the cost sensitivity of the operational mode to the relative number of new and refurbished payloads for the reusable program. Because both program examples have low unit cost satellites the advantage of reuse is less pronounced.

Comparisons and selections of program approaches are based on relative total program cost estimates required for the operational period of 1978 through 1990. Estimates were obtained using current baseline procedures and have not assumed big dumb payload effects. Reliability effects owing to launch vehicle failures are not included in the costs.

7.3.1.1 Comparison of Operational Modes for Tracking and Data Relay Satellite (TDRS) Programs

Three TDRS programs associated with satellite configurations B, C, and D were considered for meeting an increase in traffic demand by an expendable and reusable operational mode. Costs associated with these approaches over an operating period between 1978 and 1990 are compared in Table 7-34. Configuration B, called the baseline reusable, is a typical payload optimized for an expendable launch vehicle and with minimum modifications for launching with the Shuttle and Tug. Configuration C is an optimized reusable design and configuration D is a shorter version of C for possible multiple launches (see Section 5 for descriptions of configurations A through D). For all cases, configuration D has the lowest program cost. Numbers in parenthesis are estimates of program costs with multiple launches made whenever possible, considering opportunities for multiple launches for configuration D with itself.

Cases 1, 3, 4, and 5 represent various alternate approaches for structuring the payload traffic to meet the system demand. (See Section 5 and Table 5-1.) Case 1 is an expendable mode, while Cases 3, 4, and 5 are for a reusable mode. The increasing demand is satisfied by periodic payload R&D for Case 1, only mission equipment R&D for Case 3, and increases in satellite on-orbit traffic for Cases 4 and 5. Case 4 has twice and Case 5 has triple the original demand after 12 years. In the cases examined configuration D in the reusable mode provides the lowest

Table 7-34. TDRS Direct Program Cost Estimate Comparison of Operational Modes to Meet Increasing Demand (Millions of 1971 Dollars)

Cases	1	3	4	5
Operational Mode	Expendable	Reusable		
Program Changes	Payload R&D	Mission Eq. R&D	Demand X 2	Demand X 3
Configuration				
B. Baseline reusable	362	N/A*	308	404
C. Optimized reusable	361	N/A*	258	347
D. Optimized reusable, short configuration	357	274	249	344
(multiple launch of D)	(335)	(252)	(227)	(311)

*new spacecraft required

cost. Among the reusable cases the lowest cost approach for meeting the demand is dependent on the rate of demand increase. For an increase of demand by a factor of 2 the low cost approach is to use more of the same satellites (Case 4); for an increase by a factor of 3, however, the low cost approach is to update the mission equipment with periodic redesigns to increase the performance.

The relative cost of providing the capability to meet the increased demand is shown in Table 7-35 for reusable satellites. Case 2 represents the baseline program with the same satellite in use and constant demand. Based on estimates for configuration D, a 100-percent increase in demand (Case 4) is obtained with an 11 to 12-percent increase in program costs. In order to obtain a 200-percent increase in demand, either a 22 to 25-percent increase in program costs is required for mission equipment R&D (Case 3), or a 54-percent increase in program cost by using more of the same satellites.

A program factor not considered in the above comparisons is the cost of a special docking system required for retrieval of all the spinning satellites. The estimate for adapting the spin capability to the standard Apollo docking mechanism is $2.75 million for RDT&E and $0.66 million for each unit. These costs can be shared by all the applicable reusable satellite programs or paid for by the Tug program.

7.3.1.2 Comparison of Operational Modes for a System Test Satellite Program

Program direct cost comparisons for various approaches of conducting a System Test Satellite demonstration program are displayed in Table 7-36. All configurations are launched on the Shuttle with either expendable or reusable upper stages. Configuration A is the only expendable payload

Table 7-35. TDRS Direct Program Cost Estimate Impact of Increasing Demand for a Reusable Payload (Millions of 1971 Dollars)

Cases	2	3	4	5
Operational Mode		Reusable		
Program Changes	None	Mission Eq. R&D	Demand X 2	Demand X 3
Configuration				
B. Baseline reusable	243	N/A*	308	404
C. Optimized reusable	228	N/A*	258	347
D. Optimized reusable, short configuration	224	274	249	344
(multiple launch of D)	(202)	(252)	(227)	(311)

*new spacecraft required

Table 7-36. System Test Satellite Program Cost Estimate Comparison of Operational Mode (Millions of 1971 Dollars)

	Number of Payloads		Average Unit Cost	Payload Total	Program Direct Cost
	New	Refurb.			
Configuration					
A. Baseline expendable	8	0	8.9	134	184
B. Baseline reusable	6	2	9.3	128	194
C. Optimized reusable	6	2	10.8	141	211
D. Optimized reusable, short	6	2	10.6	140	210

case and it has the lowest program costs. Reuse does not pay off because of the few refurbishments conducted in comparison with the new satellites bought. The fact that the unit costs are low is also a contributing factor in reuse not paying off.

7.3.2 TDRS Program Characteristics and Results

The TDRS program was a demonstration of system traffic for a communication program. Different operational modes and satellite configurations were considered in order to meet various potential user demands. The analysis considered the basic number of launches required and expected number of satellite random failures in order to maintain three satellites in orbit for the basic case of constant demand (Case 2). Increases in communication traffic demand were met by periodic replacement with new satellites (Case 1) or only mission equipment upgrade (Case 3), and by increasing the number of identical satellites in orbit. The details of the cost analysis are presented in the following paragraphs.

The direct program costs for the operational period considered (1978 to 1990) are shown in Table 7-37 for configurations B, C, and D, and for all five cases. Cases are identified by the notation TDRS-1, TDRS-2, etc. These costs represent the total of the payload program and the launch vehicle direct charges. Cost streams for all these totals are shown in Table 7-38. A breakdown of the payload program costs into RDT&E, investment, and operations is shown in Table 7-39 along with the approximate number of new and refurbished payloads used during the program. The quantities are rounded off to the nearest whole number, whereas the costs are based on the fractional payloads required. From the quantities shown one can see that Case 1 is an expendable operational mode in which the payloads launched by the Shuttle and Tug are not

Table 7-37. Program Direct Cost Summary - Direct Program Costs (1 of 5)
(Millions of 1971 Dollars)

	SPACE PAYLOAD TOTAL	TRANSPORTATION LNCH VEH DIRECT	SYSTEM PROGRAM DIRECT
TDRS 1 CONF. B B/L REUSEABLE - 6 YR MMD TDRS 1 CONF. B	259.	103.	362.
TDRS 1 CONF. C OPTI. REUSEABLE - 7 YR MMD TDRS 1 CONF. C	274.	87.	361.
TDRS 1 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD TDRS 1 CONF. D	273.	87.	357.

Table 7-37. Program Direct Cost Summary - Direct Program Costs (2 of 5)
(Millions of 1971 Dollars)

	SPACE TRANSPORTATION SYSTEM PAYLOAD TOTAL	LNCH VEH DIRECT	PROGRAM DIRECT
TDRS 2 CONF. 3 P/L REUSEABLE - 6 YR MMD			
TDRS 2 CONF. 3	140.	103.	243.
TDRS 2 CONF. C OPTI. REUSEABLE - 7 YR MMD			
TDRS 2 CONF. C	141.	87.	228.
TDRS 2 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD			
TDRS 2 CONF. D	137.	87.	224.

Table 7-37. Program Direct Cost Summary - Direct Program Costs (3 of 5)
(Millions of 1971 Dollars)

	SPACE TRANSPORTATION SYSTEM PAYLOAD TOTAL	LNCH VEH DIRECT	PROGRAM DIRECT
TDRS 3 CONF. B B/L REUSEABLE - 6 YR MMD TDRS 3 CONF. B	178.	103.	281.
TDRS 3 CONF. C OPTI. REUSEABLE - 7 YR MMD TDRS 3 CONF. C	179.	101.	280.
TDRS 3 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD TDRS 3 CONF. D	173.	101.	274.

Table 7-37. Program Direct Cost Summary - Direct Program Costs (4 of 5)
(Millions of 1971 Dollars)

	SPACE TRANSPORTATION SYSTEM PAYLOAD TOTAL	LNCH VEH DIRECT	PROGRAM DIRECT
TDRS 4 CONF. B B/L REUSEABLE - 6 YR MMD			
TDRS 4 CONF. B	172.	136.	308.
TDRS 4 CONF. C OPTI. REUSEABLE - 7 YR MMU			
TDRS 4 CONF. C	161.	97.	258.
TDRS 4 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD			
TDRS 4 CONF. D	156.	93.	249.

Table 7-37. Program Direct Cost Summary - Direct Program Costs (5 of 5)
(Millions of 1971 Dollars)

	SPACE TRANSPORTATION SYSTEM PAYLOAD TOTAL	LNCH VEH DIRECT	PROGRAM DIRECT
TDRS 5 CONF. B B/L REUSEABLE - 6 YR MMD			
TDRS 5 CONF. B	221.	183.	404.
TDRS 5 CONF. C OPTI. REUSEABLE - 7 YR MMD			
TDRS 5 CONF. C	209.	138.	347.
TDRS 5 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD			
TDRS 5 CONF. D	206.	138.	344.

Table 7-38. Program Direct Cost Summary - Direct Program Costs
Space Transportation System (1 of 5)
(Millions of 1971 Dollars)

	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991
TDRS 1 CONF. B B/L REUSEABLE - 6 YR MMD																		
TDRS 1 CONF. B	0.	0.	23.	56.	22.	4.	11.	19.	44.	52.	6.	2.	12.	19.	44.	48.	0.	0.
TDRS 1 CONF. C OPTI. REUSEABLE - 7 YR MMD																		
TDRS 1 CONF. C	0.	0.	26.	61.	22.	1.	2.	21.	49.	52.	1.	1.	3.	21.	49.	52.	0.	0.
TDRS 1 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD																		
TDRS 1 CONF. D	0.	0.	31.	59.	21.	1.	2.	21.	47.	51.	1.	1.	3.	21.	47.	51.	0.	0.

Table 7-38. Program Direct Cost Summary - Direct Program Costs, Space Transportation System (2 of 5) (Millions of 1971 Dollars)

	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991
TDRS 2 CONF. B P/L REUSEABLE - 7 YR MMD																		
TDRS 2 CONF. B	0.	0.	25.	54.	22.	4.	11.	0.	2.	10.	42.	6.	14.	3.	43.	7.	0.	0.
TDRS 2 CONF. C OPTI. REUSEABLE - 7 YR MMD																		
TDRS 2 CONF. C	0.	0.	26.	59.	22.	1.	1.	2.	0.	3.	12.	42.	1.	6.	45.	8.	0.	0.
TDRS 2 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD																		
TDRS 2 CONF. D	0.	0.	26.	56.	21.	1.	1.	2.	0.	3.	12.	42.	1.	6.	45.	8.	0.	0.

Table 7-38. Program Direct Cost Summary - Direct Program Costs, Space Transportation System (3 of 5) (Millions of 1971 Dollars)

	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991
TDRS 3 CONF. B B/L REUSEABLE - 6 YR MMD																		
TDRS 3 CONF. B	0.	0.	23.	54.	20.	2.	4.	17.	22.	46.	2.	6.	14.	6.	21.	44.	0.	0.
TDRS 3 CONF. C OPTI. REUSEABLE - 7 YR MMD																		
TDRS 3 CONF. C	0.	0.	32.	68.	25.	1.	1.	9.	23.	47.	1.	1.	3.	7.	18.	44.	0.	0.
TDRS 3 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD																		
TDRS 3 CONF. D	0.	0.	30.	65.	24.	1.	1.	9.	23.	47.	1.	1.	3.	6.	19.	44.	0.	0.

Table 7-38. Program Direct Cost Summary - Direct Program Costs, Space Transportation System (4 of 5) (Millions of 1971 Dollars)

	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991
TDRS 4 CONF. B O/L REUSEABLE - 6 YR MMD																		
TDRS 4 CONF. B	0.	0.	23.	55.	20.	2.	4.	13.	8.	17.	40.	2.	6.	25.	14.	40.	39.	0.
TDRS 4 CONF. C OPTI. REUSEABLE - 7 YR MMD																		
TDRS 4 CONF. C	0.	0.	30.	68.	25.	1.	1.	5.	6.	11.	12.	42.	2.	7.	12.	23.	13.	0.
TDRS 4 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD																		
TDRS 4 CONF. D	0.	0.	28.	66.	24.	1.	1.	5.	6.	10.	11.	41.	2.	7.	12.	22.	13.	0.

Table 7-38. Program Direct Cost Summary - Direct Program Costs, Space Transportation System (5 of 5) (Millions of 1971 Dollars)

	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991
TDRS 5 CONF. B B/L REUSEABLE - 6 YR MMD																		
TDRS 5 CONF. B	0.	0.	23.	54.	20.	2.	4.	18.	19.	33.	41.	5.	12.	36.	23.	74.	40.	0.
TDRS 5 CONF. C OPTI. REUSEABLE - 7 YR MMD																		
TDRS 5 CONF. C	0.	0.	29.	66.	25.	1.	1.	11.	19.	28.	12.	43.	3.	13.	19.	36.	41.	0.
TDRS 5 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD																		
TDRS 5 CONF. D	0.	0.	29.	65.	24.	1.	1.	11.	19.	27.	12.	43.	3.	13.	19.	35.	42.	0.

Table 7-39. TDRS Case (1 of 5)

		PAYLOAD TOTALS					
PAYLOAD	QUANTITIES			PAYLOAD PROGRAM COSTS			
	TYPE	NEW	REFURBED	RDTE	INVEST	OPS	TOTAL
TDRS 1 CONF. B B/L REUSEABLE - 6 YR MMD							
TDRS 1 CONF. B	CURR REUSABLE	11.	0.	145.	105.	9.	259.
TDRS 1 CONF. C OPTI. REUSEABLE - 7 YR MMD							
TDRS 1 CONF. C	CURR REUSABLE	9.	0.	159.	106.	9.	274.
TDRS 1 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD							
TDRS 1 CONF. D	CURR REUSABLE	9.	0.	156.	105.	9.	270.

Table 7-39. TDRS Case (2 of 5)

	PAYLOAD TOTALS						
	PAYLOAD QUANTITIES			PAYLOAD PROGRAM COSTS			
	TYPE	NEW	REFURBED	RDTE	INVEST	OPS	TOTAL
TDRS 2 CONF. 3 L/L REUSEABLE - 6 YR MMD							
TDRS 2 CONF. 3	CURR REUSABLE	7.	4.	49.	68.	23.	140.
TDRS 2 CONF. 3 OPTI. REUSEABLE - 7 YR MMD							
TDRS 2 CONF. 3	CURR REUSABLE	5.	4.	54.	61.	26.	141.
TDRS 2 CONF. 3 OPTI. REUSEABLE SHORT - 7 YR MMD							
TDRS 2 CONF. 3	CURR REUSABLE	5.	4.	52.	59.	26.	137.

Table 7-39. TDRS Case (3 of 5)

PAYLOAD	TYPE	PAYLOAD TOTALS QUANTITIES NEW	REFURBED	PAYLOAD PROGRAM COSTS RDTE	INVEST	OPS	TOTAL
TDRS 3 CONF. B C/L REUSEABLE - 6 YR MMD							
TDRS 3 CONF. B	CURR REUSABLE	7.	4.	75.	68.	35.	178.
TDRS 3 CONF. C OPTI. REUSEABLE - 7 YR MMD							
TDRS 3 CONF. C	CURR REUSABLE	5.	4.	81.	61.	37.	179.
TDRS 3 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD							
TDRS 3 CONF. D	CURR REUSABLE	5.	4.	78.	59.	36.	173.

Table 7-39. TDRS Case (4 of 5)

PAYLOAD	TYPE	PAYLOAD TOTALS QUANTITIES NEW	REFURBED	PAYLOAD PROGRAM COSTS RDTE	INVEST	OPS	TOTAL
TDRS 4 CONF. B E/L REUSEABLE - 6 YR MMD							
TDRS 4 CONF. B	CURR REUSABLE	9.	6.	49.	88.	35.	172.
TDRS 4 CONF. C OPTI. REUSEABLE - 7 YR MMD							
TDRS 4 CONF. C	CURR REUSABLE	7.	3.	54.	84.	23.	161.
TDRS 4 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD							
TDRS 4 CONF. D	CURR REUSABLE	7.	3.	52.	82.	22.	156.

Table 7-39. TDRS Case (5 of 5)

	PAYLOAD TYPE	PAYLOAD TOTALS PAYLOAD QUANTITIES NEW	REFURBED	PAYLOAD PROGRAM COSTS RDTE	INVEST	OPS	TOTAL
TDRS 5 CONF. B B/L REUSEABLE - 6 YR MMD							
TDRS 5 CONF. B	CURR REUSABLE	13.	8.	49.	125.	47.	221.
TDRS 5 CONF. C OPTI. REUSEABLE - 7 YR MMD							
TDRS 5 CONF. C	CURR REUSABLE	11.	5.	54.	119.	36.	209.
TDRS 5 CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD							
TDRS 5 CONF. D	CURR REUSABLE	11.	5.	52.	118.	36.	206.

recovered even though they are retrievable designs. Configuration B is a baseline (B/L) reusable design which is a minimum modification to an expendable design optimized for an expendable launch vehicle. The modification permits the payload to be attached to the Tug and launched from the Shuttle. Configuration C is a life-optimized design for reuse, and configuration D is a life-optimized design which was shortened to permit at least two satellites to be launched with the Tug on a single Shuttle flight.

The Intelsat IV satellite designs are used in this study as an example of the TDRS configurations. Characteristics of the designs used for costing are displayed in Table 7-40 for configurations B, C, and D. Basic satellites costs as well as program costs are broken out for all of the 13 combinations of case and configuration. The estimates are approximations in that the redundancy cost factors of 6-year MMD (configuration B) and 7-year MMD (configurations C and D) satellites are from the baseline CERs.

Details of the satellite programs are shown by the schedule and quantity for new designs and for launches. Included in the schedule are launches of new, refurbished, and replacement (for random failure) payloads. Corresponding spread cost streams are displayed at the bottom of the tables. Reliability effects owing to launch vehicle failure were not included in this estimate.

Launch schedules and direct charges based on a capture analysis are presented in Table 7-41 for all 13 individual programs. In the first year of the program, 1978, expendable launch vehicles were required for the reusable type payloads. For this exercise additional costs were not included for adapting these payloads to the expendable launch vehicle. Launch vehicles considered were the Titan IIID/Centaur and the Titan IIIC. From 1978 on, Shuttle and reusable Tugs were used. The estimated costs for the launches include trip sharing with other programs where possible.

Table 7-40. (1 of 15) TDRS 1 Configuration B

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE	WEIGHTS				REDUNDANT COST FACTOR		BASIC	AVG	FIRST	PAYLOAD PROGRAM COST ESTIMATE			
SUBSYSTEM	DRY	TOTAL	OTHER INPUTS		DEV	PROD	RDTE	UNIT	UNIT	RDTE	INVEST	OPS	TOTAL
STRUCTURE	619	619	TYPE,	END	1.000	1.000	9.7	1.4	1.6	29.	16.	0.	45.
ELECTRICAL POWER	259	259	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	24.	13.	0.	37.
TRACKING,COMMAND	50	50	ALT,	SYNC	1.000	1.000	6.5	1.1	1.2	19.	13.	0.	32.
STABILITY,CONTROL	91	437	TYPE,	SPIN	1.000	1.000	5.9	1.2	1.3	18.	13.	0.	31.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1019	1365					30.0	5.1	5.5	90.	55.	0.	145.
MISSION EQUIPMENT	350	350	COMPLXTY,	LOW	1.000	1.000	12.5	4.2	4.6	37.	46.	0.	83.
SATELLITE	1369	1715					42.5	9.3	10.1	127.	101.	0.	228.
AGE					1.000		2.9			9.	0.	0.	9.
LAUNCH SUPPORT						1.000		.8	.8	0.	0.	9.	9.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	2.	0.	3.
SE AND TD										8.	2.	0.	10.
TOTAL										145.	105.	9.	259.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00					1.00						1.00		3.0
MISSION EQUIPMENT	1.00					1.00						1.00		3.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	1.	0.	0.	3.	0.	1.	0.	0.	0.	3.	0.	11.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	12.	27.	10.	0.	0.	12.	27.	10.	0.	0.	0.	12.	27.	8.	0.	145.
INVESTMENT	0.	0.	7.	16.	8.	4.	2.	7.	16.	8.	6.	2.	0.	7.	16.	6.	0.	105.
OPERATIONS	0.	0.	0.	4.	1.	0.	0.	0.	1.	1.	0.	0.	0.	0.	1.	1.	0.	9.
TOTAL	0.	0.	19.	47.	19.	4.	2.	19.	44.	19.	6.	2.	0.	19.	44.	15.	0.	259.

Table 7-40. (2 of 15) TDRS 1 Configuration C

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE	WEIGHTS				REDUNDANT COST FACTOR		BASIC	AVG	FIRST	PAYLOAD PROGRAM COST ESTIMATE			
SUBSYSTEM	DRY	TOTAL	OTHER INPUTS		DEV	PROD	RDTE	UNIT	UNIT	RDTE	INVEST	OPS	TOTAL
STRUCTURE	554	684	TYPE,	END	1.000	1.000	10.0	1.5	1.7	30.	14.	0.	44.
ELECTRICAL POWER	380	380	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	24.	12.	0.	36.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	25.	18.	0.	43.
STABILITY,CONTROL	145	981	TYPE,	SPIN	1.000	1.000	7.5	1.8	2.0	23.	17.	0.	40.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1286	2122					33.7	6.5	7.0	102.	61.	0.	163.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	38.	41.	0.	79.
SATELLITE	1654	2490					46.4	10.8	11.7	140.	102.	0.	242.
AGE					1.000		2.9			9.	0.	0.	9.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	9.	9.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	2.	0.	3.
SE AND TD										9.	2.	0.	11.
TOTAL										159.	106.	9.	274.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00					1.00						1.00		3.0
MISSION EQUIPMENT	1.00					1.00						1.00		3.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	0.	0.	3.	0.	0.	0.	0.	0.	3.	0.	9.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	13.	29.	11.	0.	0.	13.	29.	11.	0.	0.	0.	13.	29.	11.	0.	159.
INVESTMENT	0.	0.	9.	19.	7.	1.	0.	8.	19.	7.	1.	1.	0.	8.	19.	7.	0.	106.
OPERATIONS	0.	0.	0.	4.	1.	0.	0.	0.	1.	1.	0.	0.	0.	0.	1.	1.	0.	9.
TOTAL	0.	0.	22.	52.	19.	1.	0.	21.	49.	19.	1.	1.	0.	21.	49.	19.	0.	274.

Table 7-40. (3 of 15) TDRS 1 Configuration D

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	580	580	TYPE,	END	1.000	1.000	9.5	1.4	1.5	28.	13.	0.	41.
ELECTRICAL POWER	380	380	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	24.	12.	0.	36.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	25.	18.	0.	43.
STABILITY,CONTROL	145	929	TYPE,	SPIN	1.000	1.000	7.4	1.8	1.9	22.	17.	0.	39.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1162	1966					33.0	6.3	6.8	99.	60.	0.	159.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	38.	41.	0.	79.
SATELLITE	1550	2334					45.7	10.6	11.5	137.	101.	0.	238.
AGE					1.000		2.9			9.	0.	0.	9.
LAUNCH SUPPORT						1.300		1.0	1.0	0.	0.	9.	9.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	2.	0.	3.
SE AND TD										9.	2.	0.	11.
TOTAL										156.	105.	9.	270.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00					1.00						1.00		3.0
MISSION EQUIPMENT	1.00					1.00						1.00		3.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	0.	0.	3.	0.	0.	0.	0.	0.	3.	0.	9.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	16.	28.	10.	0.	0.	13.	28.	10.	0.	0.	0.	13.	28.	10.	0.	156.
INVESTMENT	0.	0.	11.	18.	7.	1.	0.	8.	18.	7.	1.	1.	0.	8.	18.	7.	0.	105.
OPERATIONS	0.	0.	0.	4.	1.	0.	0.	0.	1.	1.	0.	0.	0.	0.	1.	1.	0.	9.
TOTAL	0.	0.	27.	50.	18.	1.	0.	21.	47.	18.	1.	1.	0.	21.	47.	18.	0.	270.

Table 7-40. (4 of 15) TDRS 2 Configuration B

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE	WEIGHTS		OTHER INPUTS		COST FACTOR		BASIC	AVG	FIRST	PAYLOAD PROGRAM COST ESTIMATE			
SUBSYSTEM	DRY	TOTAL			DEV	PROD	RDTE	UNIT	UNIT	RDTE	INVEST	OPS	TOTAL
STRUCTURE	619	619	TYPE,	END	1.000	1.000	9.7	1.4	1.6	10.	10.	0.	20.
ELECTRICAL POWER	259	259	WATTS,	57..	1.000	1.000	7.9	1.2	1.3	8.	9.	0.	17.
TRACKING,COMMAND	50	50	ALT,	SYNC	1.000	1.000	6.5	1.1	1.2	6.	8.	0.	14.
STABILITY,CONTROL	91	437	TYPE,	SPIN	1.000	1.000	5.9	1.2	1.3	6.	9.	0.	15.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1019	1365					30.0	5.1	5.5	30.	36.	0.	66.
MISSION EQUIPMENT	350	350	COMPLXTY,	LOW	1.000	1.000	12.5	4.2	4.6	12.	29.	0.	41.
SATELLITE	1369	1715					42.5	9.3	10.1	42.	65.	14.	121.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		.8	.8	0.	0.	9.	9.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										3.	2.	0.	5.
TOTAL										49.	68.	23.	140.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00													1.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	1.	0.	0.	0.	1.	0.	1.	0.	0.	1.	0.	7.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	2.	0.	0.	0.	0.	2.	0.	4.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	12.	27.	10.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	49.
INVESTMENT	0.	0.	9.	16.	8.	4.	2.	0.	2.	5.	4.	6.	2.	3.	5.	2.	0.	68.
OPERATIONS	0.	0.	0.	2.	1.	0.	0.	0.	0.	5.	5.	0.	0.	0.	5.	5.	0.	23.
TOTAL	0.	0.	21.	45.	19.	4.	2.	0.	2.	10.	9.	6.	2.	3.	10.	7.	0.	140.

Table 7-40. (5 of 15) TDRS 2 Configuration C

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	634	654	TYPE,	END	1.000	1.000	10.0	1.5	1.7	10.	8.	0.	18.
ELECTRICAL POWER	380	380	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	7.	0.	15.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	8.	10.	0.	18.
STABILITY,CONTROL	145	981	TYPE,	SPIN	1.000	1.000	7.5	1.8	2.0	8.	10.	0.	18.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1246	2122					33.7	6.5	7.0	34.	35.	0.	69.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	13.	23.	0.	36.
SATELLITE	1554	2490					46.4	10.8	11.7	47.	58.	17.	122.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	9.	9.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										3.	2.	0.	5.
TOTAL										54.	61.	26.	141.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00													1.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	0.	0.	0.	0.	1.	0.	0.	0.	1.	0.	5.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	0.	2.	0.	0.	0.	2.	0.	4.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	13.	30.	11.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	54.
INVESTMENT	0.	0.	9.	19.	7.	1.	1.	0.	0.	3.	6.	3.	1.	3.	6.	2.	0.	61.
OPERATIONS	0.	0.	0.	1.	1.	0.	0.	0.	0.	0.	6.	6.	0.	0.	6.	6.	0.	26.
TOTAL	0.	0.	22.	50.	19.	1.	1.	0.	0.	3.	12.	9.	1.	3.	12.	8.	0.	141.

Table 7-40. (6 of 15) TDRS 2 Configuration D

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	580	580	TYPE,	END	1.000	1.000	9.5	1.4	1.5	9.	7.	0.	16.
ELECTRICAL POWER	380	380	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	7.	0.	15.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	8.	10.	0.	18.
STABILITY,CONTROL	145	929	TYPE,	SPIN	1.000	1.000	7.4	1.8	1.9	7.	9.	0.	16.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1182	1966					33.0	6.3	6.8	32.	33.	0.	65.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	13.	23.	0.	36.
SATELLITE	1550	2334					45.7	10.6	11.5	45.	56.	17.	118.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	9.	9.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										3.	2.	0.	5.
TOTAL										52.	59.	26.	137.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00													1.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	0.	0.	0.	0.	1.	0.	0.	0.	1.	0.	5.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	0.	2.	0.	0.	0.	2.	0.	4.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	14.	28.	10.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	52.
INVESTMENT	0.	0.	8.	18.	7.	1.	1.	0.	0.	3.	6.	3.	1.	3.	6.	2.	0.	59.
OPERATIONS	0.	0.	0.	1.	1.	0.	0.	0.	0.	0.	6.	6.	0.	0.	6.	6.	0.	26.
TOTAL	0.	0.	22.	47.	18.	1.	1.	0.	0.	3.	12.	9.	1.	3.	12.	8.	0.	137.

Table 7-40. (7 of 15) TDRS 3 Configuration B

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	619	619	TYPE,	END	1.000	1.000	9.7	1.4	1.6	10.	10.	0.	20.
ELECTRICAL POWER	259	259	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	9.	0.	17.
TRACKING,COMMAND	50	50	ALT,	SYNC	1.000	1.000	6.5	1.1	1.2	6.	8.	0.	14.
STABILITY,CONTROL	91	437	TYPE,	SPIN	1.000	1.000	5.9	1.2	1.3	6.	9.	0.	15.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1019	1365					30.0	5.1	5.5	30.	36.	0.	66.
MISSION EQUIPMENT	350	350	COMPLXTY,	LOW	1.000	1.000	12.5	4.2	4.6	37.	29.	0.	66.
SATELLITE	1369	1715					42.5	9.3	10.1	67.	65.	26.	158.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		.8	.8	0.	0.	9.	9.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										4.	2.	0.	6.
TOTAL										75.	68.	35.	178.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00					1.00						1.00		3.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	1.	0.	1.	0.	0.	1.	0.	0.	1.	0.	7.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	2.	0.	0.	0.	0.	0.	2.	0.	4.
MAINTENANCE FLTS	0.000	0.000	0.000	0.000	0.000	.600	0.000	0.000	0.000	0.000	0.000	.600	0.000	1.200

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	12.	27.	10.	0.	0.	3.	7.	3.	0.	0.	0.	3.	7.	3.	0.	75.
INVESTMENT	0.	0.	7.	16.	6.	2.	4.	5.	7.	2.	2.	6.	2.	3.	6.	0.	0.	68.
OPERATIONS	0.	0.	0.	2.	1.	0.	0.	0.	8.	8.	0.	0.	0.	0.	8.	8.	0.	35.
TOTAL	0.	0.	19.	45.	17.	2.	4.	8.	22.	13.	2.	6.	2.	6.	21.	11.	0.	178.

Table 7-40. (8 of 15) TDRS 3 Configuration C

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	684	684	TYPE,	END	1.000	1.000	10.0	1.5	1.7	10.	8.	0.	18.
ELECTRICAL POWER	380	380	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	7.	0.	15.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	8.	10.	0.	18.
STABILITY,CONTROL	145	981	TYPE,	SPIN	1.000	1.000	7.5	1.8	2.0	8.	10.	0.	18.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1286	2122					33.7	6.5	7.0	34.	35.	0.	69.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	38.	23.	0.	61.
SATELLITE	1654	2490					46.4	10.8	11.7	72.	58.	28.	158.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	9.	9.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										5.	2.	0.	7.
TOTAL										81.	61.	37.	179.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS																		
SPACECRAFT					1.00													1.0
MISSION EQUIPMENT					1.00					1.00						1.00		3.0
SATELLITE SCHEDULE																		
NEW (CURR REUSABLE)					3.	0.	0.	0.	0.	1.	0.	0.	0.	0.	0.	1.	0.	5.
REFURB (RATE=.390)					0.	0.	0.	0.	0.	2.	0.	0.	0.	0.	0.	2.	0.	4.
MAINTENANCE FLTS					0.000	0.000	0.000	0.000	0.000	.500	0.000	0.000	0.000	0.000	0.000	.500	0.000	1.000
FUNDING																		
RDTE	0.	0.	15.	29.	11.	0.	0.	3.	7.	3.	0.	0.	0.	3.	7.	3.	0.	81.
INVESTMENT	0.	0.	9.	19.	7.	1.	1.	4.	8.	3.	1.	1.	0.	4.	3.	0.	0.	61.
OPERATIONS	0.	0.	0.	4.	1.	0.	0.	0.	8.	8.	0.	0.	0.	0.	8.	8.	0.	37.
TOTAL	0.	0.	24.	52.	19.	1.	1.	7.	23.	14.	1.	1.	0.	7.	18.	11.	0.	179.

Table 7-40. (9 of 15) TDRS 3 Configuration D

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	580	580	TYPE,	END	1.000	1.000	9.5	1.4	1.5	9.	7.	0.	16.
ELECTRICAL POWER	380	380	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	7.	0.	15.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	8.	10.	0.	18.
STABILITY,CONTROL	145	929	TYPE,	SPIN	1.000	1.000	7.4	1.8	1.9	7.	9.	0.	16.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1162	1966					33.0	6.3	6.8	32.	33.	0.	65.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	38.	23.	0.	61.
SATELLITE	1550	2334					45.7	10.6	11.5	57.	56.	27.	140.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	9.	9.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										4.	2.	0.	6.
TOTAL										78.	59.	36.	173.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00					1.00						1.00		3.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	0.	0.	1.	0.	0.	0.	0.	0.	1.	0.	5.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	2.	0.	0.	0.	0.	0.	2.	0.	4.
MAINTENANCE FLTS	0.000	0.000	0.000	0.000	0.000	.500	0.000	0.000	0.000	0.000	0.000	.500	0.000	1.000

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	14.	28.	10.	0.	0.	3.	7.	3.	0.	0.	0.	3.	7.	3.	0.	78.
INVESTMENT	0.	0.	8.	18.	7.	1.	1.	4.	8.	3.	1.	1.	0.	3.	4.	0.	0.	59.
OPERATIONS	0.	0.	0.	3.	1.	0.	0.	0.	8.	8.	0.	0.	0.	0.	8.	8.	0.	36.
TOTAL	0.	0.	22.	49.	18.	1.	1.	7.	23.	14.	1.	1.	0.	6.	19.	11.	0.	160.

Table 7-40. (10 of 15) TDRS 4 Configuration B

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	619	619	TYPE,	END	1.000	1.000	9.7	1.4	1.6	10.	13.	0.	23.
ELECTRICAL POWER	259	259	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	11.	0.	19.
TRACKING,COMMAND	50	50	ALT,	SYNC	1.000	1.000	6.5	1.1	1.2	6.	11.	0.	17.
STABILITY,CONTROL	91	437	TYPE,	SPIN	1.000	1.000	5.9	1.2	1.3	6.	11.	0.	17.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1019	1365					30.0	5.1	5.5	30.	46.	0.	76.
MISSION EQUIPMENT	350	350	COMPLXTY,	LOW	1.000	1.000	12.5	4.2	4.6	12.	39.	0.	51.
SATELLITE	1369	1715					42.5	9.3	10.1	42.	85.	22.	149.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		.8	.8	0.	0.	13.	13.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										3.	2.	0.	5.
TOTAL										49.	88.	35.	172.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00													1.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	1.	0.	1.	1.	0.	0.	1.	1.	1.	1.	9.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	2.	0.	0.	0.	0.	2.	2.	6.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1985	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	12.	27.	10.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	49.
INVESTMENT	0.	0.	7.	16.	6.	2.	4.	4.	8.	7.	2.	2.	6.	7.	9.	7.	1.	88.
OPERATIONS	0.	0.	0.	3.	1.	0.	0.	0.	0.	5.	5.	0.	0.	1.	5.	10.	5.	35.
TOTAL	0.	0.	19.	46.	17.	2.	4.	4.	8.	12.	7.	2.	6.	8.	14.	17.	6.	172.

Table 7-40. (11 of 15) TDRS 4 Configuration C

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC ROTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE ROTE	INVEST	OPS	TOTAL
STRUCTURE	684	684	TYPE,	END	1.000	1.000	10.0	1.5	1.7	10.	11.	0.	21.
ELECTRICAL POWER	380	380	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	9.	0.	17.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	8.	14.	0.	22.
STABILITY,CONTROL	145	981	TYPE,	SPIN	1.000	1.000	7.5	1.8	2.0	8.	14.	0.	22.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1286	2122					33.7	6.5	7.0	34.	48.	0.	82.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	13.	33.	0.	46.
SATELLITE	1654	2490					46.4	10.8	11.7	47.	81.	13.	141.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	10.	10.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										3.	2.	0.	5.
TOTAL										54.	84.	23.	161.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00													1.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	0.	0.	1.	0.	1.	0.	0.	0.	1.	1.	7.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	0.	2.	0.	0.	0.	1.	0.	3.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
ROTE	0.	0.	13.	30.	11.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	54.
INVESTMENT	0.	0.	9.	19.	7.	1.	1.	3.	6.	5.	6.	3.	2.	3.	9.	8.	2.	84.
OPERATIONS	0.	0.	0.	3.	1.	0.	0.	0.	0.	0.	6.	6.	0.	0.	3.	4.	0.	23.
TOTAL	0.	0.	22.	52.	19.	1.	1.	3.	6.	5.	12.	9.	2.	3.	12.	12.	2.	161.

Table 7-40. (12 of 15) TDRS 4 Configuration D

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	580	930	TYPE,	END	1.000	1.000	9.5	1.4	1.5	9.	10.	0.	19.
ELECTRICAL POWER	380	380	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	9.	0.	17.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	8.	14.	0.	22.
STABILITY,CONTROL	145	929	TYPE,	SPIN	1.000	1.000	7.4	1.8	1.9	7.	13.	0.	20.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1182	1966					33.0	6.3	6.8	32.	46.	0.	78.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	13.	33.	0.	46.
SATELLITE	1550	2334					45.7	10.6	11.5	45.	79.	12.	136.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	10.	10.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	1.	0.	2.
SE AND TD										3.	2.	0.	5.
TOTAL										52.	82.	22.	156.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00													1.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	0.	0.	1.	0.	1.	0.	0.	0.	1.	1.	7.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	0.	2.	0.	0.	0.	1.	0.	3.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	13.	29.	10.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	52.
INVESTMENT	0.	0.	8.	18.	7.	1.	1.	3.	6.	5.	6.	3.	2.	3.	9.	8.	2.	82.
OPERATIONS	0.	0.	0.	5.	1.	0.	0.	0.	0.	0.	5.	5.	0.	0.	3.	3.	0.	22.
TOTAL	0.	0.	21.	52.	18.	1.	1.	3.	6.	5.	11.	8.	2.	3.	12.	11.	2.	156.

Table 7-40. (13 of 15) TDRS 5 Configuration B

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	619	619	TYPE,	END	1.000	1.000	9.7	1.4	1.6	10.	19.	0.	29.
ELECTRICAL POWER	259	259	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	16.	0.	24.
TRACKING,COMMAND	50	50	ALT,	SYNC	1.000	1.000	6.5	1.1	1.2	6.	15.	0.	21.
STABILITY,CONTROL	91	437	TYPE,	SPIN	1.000	1.000	5.9	1.2	1.3	6.	16.	0.	22.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1019	1365					30.0	5.1	5.5	30.	66.	0.	96.
MISSION EQUIPMENT	350	350	COMPLXTY,	LOW	1.000	1.000	12.5	4.2	4.6	12.	55.	0.	67.
SATELLITE	1369	1715					42.5	9.3	10.1	42.	121.	29.	192.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		.8	.8	0.	0.	18.	18.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	2.	0.	3.
SE AND TD										3.	2.	0.	5.
TOTAL										49.	125.	47.	221.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00													1.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	1.	0.	3.	1.	0.	0.	2.	0.	2.	1.	13.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	2.	0.	0.	0.	0.	4.	2.	8.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	12.	27.	10.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	49.
INVESTMENT	0.	0.	7.	16.	6.	2.	4.	9.	18.	11.	2.	5.	11.	10.	13.	9.	2.	125.
OPERATIONS	0.	0.	0.	2.	1.	0.	0.	0.	1.	6.	5.	0.	1.	1.	10.	15.	5.	47.
TOTAL	0.	0.	19.	45.	17.	2.	4.	9.	19.	17.	7.	5.	12.	11.	23.	24.	7.	221.

Table 7-40. (14 of 15) TDRS 5 Configuration C

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	684	684	TYPE,	END	1.000	1.000	10.0	1.5	1.7	10.	16.	0.	26.
ELECTRICAL POWER	380	330	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	13.	0.	21.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	8.	21.	0.	29.
STABILITY,CONTROL	145	981	TYPE,	SPIN	1.000	1.000	7.5	1.8	2.0	8.	19.	0.	27.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1286	2122					33.7	6.5	7.0	34.	69.	0.	103.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	13.	46.	0.	59.
SATELLITE	1654	2490					46.4	10.8	11.7	47.	115.	21.	183.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	15.	15.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	2.	0.	3.
SE AND TD										3.	2.	0.	5.
TOTAL										54.	119.	36.	209.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00													1.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	0.	0.	3.	0.	1.	0.	1.	0.	2.	1.	11.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	0.	2.	0.	0.	0.	1.	2.	5.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	13.	30.	11.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	54.
INVESTMENT	0.	0.	8.	18.	7.	1.	1.	9.	18.	10.	6.	4.	3.	7.	15.	11.	1.	119.
OPERATIONS	0.	0.	0.	2.	1.	0.	0.	0.	1.	1.	6.	6.	0.	0.	4.	9.	6.	36.
TOTAL	0.	0.	21.	50.	19.	1.	1.	9.	19.	11.	12.	10.	3.	7.	19.	20.	7.	209.

Table 7-40. (15 of 15) TDRS 5 Configuration D

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	580	580	TYPE,	END	1.000	1.000	9.5	1.4	1.5	9.	15.	0.	24.
ELECTRICAL POWER	383	383	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	8.	13.	0.	21.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	8.	21.	0.	29.
STABILITY,CONTROL	145	929	TYPE,	SPIN	1.000	1.000	7.4	1.8	1.9	7.	19.	0.	26.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1182	1966					33.0	6.3	6.8	32.	68.	0.	100.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	13.	46.	0.	59.
SATELLITE	1550	2334					45.7	10.6	11.5	45.	114.	21.	180.
AGE					1.000		2.9			3.	0.	0.	3.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	15.	15.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	2.	0.	3.
SE AND TD										3.	2.	0.	5.
TOTAL										52.	118.	36.	206.

FISCAL YEAR	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00													1.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	3.	0.	0.	0.	0.	3.	0.	1.	0.	1.	0.	2.	1.	11.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	0.	2.	0.	0.	0.	1.	2.	5.

FISCAL YEAR	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	TOTAL
FUNDING																		
RDTE	0.	0.	13.	29.	10.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	52.
INVESTMENT	0.	0.	8.	18.	7.	1.	1.	9.	18.	9.	6.	4.	3.	7.	15.	10.	2.	118.
OPERATIONS	0.	0.	0.	2.	1.	0.	0.	0.	1.	1.	6.	6.	0.	0.	4.	9.	6.	36.
TOTAL	0.	0.	21.	49.	18.	1.	1.	9.	19.	10.	12.	10.	3.	7.	19.	19.	8.	206.

Table 7-41. (1 of 15) Individual Program Cost Breakdown - TDRS 1 Configuration B (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES LAUNCH VEHICLES T3D/C				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	4.	23.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	47.	9.	56.
1978	3.	0.	1.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	3.	22.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.	0.	4.
1980	1.	0.	0.0	0.0	0.0	0.0	.8	.8	0.0	2.	9.	11.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	0.	19.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	44.	0.	44.
1983	3.	0.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	19.	33.	52.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	0.	6.
1985	1.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	2.	0.	2.
1986	0.	0.	0.0	0.0	0.0	0.0	1.1	1.1	0.0	0.	12.	12.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	0.	19.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	44.	0.	44.
1989	3.	0.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	15.	33.	48.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	11.	0.	1.0	0.0	0.0	0.0	7.8	7.8	0.0	259.	103.	362.

Table 7-41. (2 of 15) Individual Program Cost Breakdown - TDRS 1 Configuration C (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES PAYLOADS NEW	REFURB	LAUNCH VEHICLES T3D/C				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	22.	4.	26.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	52.	9.	61.
1978	3.	0.	1.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	3.	22.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1980	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	0.	2.	2.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	21.	0.	21.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	49.	0.	49.
1983	3.	0.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	19.	33.	52.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1986	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	0.	3.	3.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	21.	0.	21.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	49.	0.	49.
1989	3.	0.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	19.	33.	52.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	9.	0.	1.0	0.0	0.0	0.0	6.4	6.4	0.0	274.	87.	361.

Table 7-41. (3 of 15) Individual Program Cost Breakdown - TDRS 1 Configuration D (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	T3D/C				SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES		LAUNCH VEHICLES							PROGRAM DIRECT COST		
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	27.	4.	31.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	50.	9.	59.
1978	3.	0.	1.0	0.0	0.0	0.0	0.0	0.0	0.0	18.	3.	21.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1980	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	0.	2.	2.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	21.	0.	21.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	47.	0.	47.
1983	3.	0.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	18.	33.	51.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1986	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	0.	3.	3.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	21.	0.	21.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	47.	0.	47.
1989	3.	0.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	18.	33.	51.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	9.	0.	1.0	0.0	0.0	0.0	6.4	6.4	0.0	270.	87.	357.

Table 7-41. (4 of 15) Individual Program Cost Breakdown - TDRS 2 Configuration B (Millions of 1971 Dollars)

LAUNCHED FROM ETR

	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES										PROGRAM DIRECT COST		
			LAUNCH VEHICLES										
FISCAL YEAR	PAYLOADS NEW	REFURB	T3D/C				SHTL SHTL	TUG	TUG EXP		PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		21.	4.	25.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		45.	9.	54.
1978	3.	0.	1.0	0.0	0.0	0.0	0.0	0.0	0.0		19.	3.	22.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		4.	0.	4.
1980	1.	0.	0.0	0.0	0.0	0.0	.8	.8	0.0		2.	9.	11.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		2.	0.	2.
1983	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		10.	0.	10.
1984	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0		9.	33.	42.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		6.	0.	6.
1986	1.	0.	0.0	0.0	0.0	0.0	1.1	1.1	0.0		2.	12.	14.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		3.	0.	3.
1988	0.	0.	0.0	0.0	0.0	0.0	3.0	3.0	0.0		10.	33.	43.
1989	1.	2.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		7.	0.	7.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
TOTAL	7.	4.	1.0	0.0	0.0	0.0	7.8	7.8	0.0		140.	103.	243.

Table 7-41. (5 of 15) Individual Program Cost Breakdown - TDRS 2 Configuration C (Millions of 1971 Dollars)

LAUNCHED FROM ETR

SPACE TRANSPORTATION SYSTEM

FISCAL YEAR	SCHEDULED QUANTITIES PAYLOADS NEW	REFURB	LAUNCH VEHICLES T3D/C				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	22.	4.	26.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	50.	9.	59.
1978	3.	0.	1.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	3.	22.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1981	1.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	0.	2.	2.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1983	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	3.	0.	3.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	12.	0.	12.
1985	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	9.	33.	42.
1986	0.	9.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1987	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	3.	3.	6.
1988	0.	0.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	12.	33.	45.
1989	1.	2.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	8.	0.	8.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	5.	4.	1.0	0.0	0.0	0.0	6.4	6.4	0.0	141.	87.	228.

Table 7-41. (6 of 15) Individual Program Cost Breakdown - TDRS 2 Configuration D (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES PAYLOADS NEW	REFURB.	LAUNCH VEHICLES T3D/C				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	22.	4.	26.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	47.	9.	56.
1978	3.	0.	1.0	0.0	0.0	0.0	0.0	0.0	0.0	18.	3.	21.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1981	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	0.	2.	2.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1983	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	3.	0.	3.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	12.	0.	12.
1985	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	9.	33.	42.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1987	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	3.	3.	6.
1988	0.	0.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	12.	33.	45.
1989	1.	2.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	8.	0.	8.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	5.	4.	1.0	0.0	0.0	0.0	6.4	6.4	0.0	137.	87.	224.

Table 7-41. (7 of 15) Individual Program Cost Breakdown - TDRS 3 Configuration B (Millions of 1971 Dollars)

LAUNCHED FROM ETR

	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES									PROGRAM DIRECT COST		
			LAUNCH VEHICLES									
FISCAL YEAR	PAYLOADS NEW	REFURB	T3D/ C				SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	4.	23.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	45.	9.	54.
1978	3.	0.	1.0	0.0	0.0	0.0	0.0	0.0	0.0	17.	3.	20.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	2.	0.	2.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.	0.	4.
1981	1.	0.	0.0	0.0	0.0	0.0	.8	.8	0.0	8.	9.	17.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	22.	0.	22.
1983	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	13.	33.	46.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	2.	0.	2.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	0.	6.
1986	1.	3.	0.0	0.0	0.0	0.0	1.1	1.1	0.0	2.	12.	14.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	0.	6.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	21.	0.	21.
1989	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	11.	33.	44.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	7.	4.	1.0	0.0	0.0	0.0	7.8	7.8	0.0	178.	103.	281.

Table 7-41. (8 of 15) Individual Program Cost Breakdown - TDRS 3 Configuration C (Millions of 1971 Dollars)

LAUNCHED FROM EIR

FISCAL YEAR	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES PAYLOADS NEW	REFURB	LAUNCH VEHICLES T3D/C		T3C		SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	24.	8.	32.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	52.	16.	68.
1978	3.	0.	1.0	0.0	1.0	0.0	0.0	0.0	0.0	19.	6.	25.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1981	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	7.	2.	9.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	23.	0.	23.
1983	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	14.	33.	47.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1986	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	0.	3.	3.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	7.	0.	7.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	18.	0.	18.
1989	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	11.	33.	44.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	5.	4.	1.0	0.0	1.0	0.0	6.4	6.4	0.0	179.	101.	280.

Table 7-41. (9 of 15) Individual Program Cost Breakdown - TDRS 3 Configuration D
(Millions of 1971 Dollars)

LAUNCHED FROM ETR

	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES		LAUNCH VEHICLES							PROGRAM DIRECT COST		
FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	T3D/C		T3C		SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	22.	8.	30.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	49.	16.	65.
1978	3.	0.	1.0	0.0	1.0	0.0	0.0	0.0	0.0	18.	6.	24.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1981	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	7.	2.	9.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	23.	0.	23.
1983	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	14.	33.	47.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1986	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	0.	3.	3.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	3.	0.	3.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	12.	0.	12.
1989	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	8.	33.	41.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	5.	4.	1.0	0.0	1.0	0.0	6.4	6.4	0.0	160.	101.	261.

Table 7-41. (10 of 15) Individual Program Cost Breakdown - TDRS 4 Configuration B (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES LAUNCH VEHICLES T3D/C				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	4.	23.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	46.	9.	55.
1978	3.	0.	1.0	0.0	0.0	0.0	0.0	0.0	0.0	17.	3.	20.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	2.	0.	2.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.	0.	4.
1981	1.	0.	0.0	0.0	0.0	0.0	.8	.8	0.0	4.	9.	13.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	8.	0.	8.
1983	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	12.	5.	17.
1984	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	7.	33.	40.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	2.	0.	2.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	0.	6.
1987	1.	0.	0.0	0.0	0.0	0.0	1.5	1.5	0.0	8.	17.	25.
1988	1.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	14.	0.	14.
1989	1.	2.	0.0	0.0	0.0	0.0	2.0	2.0	0.0	17.	23.	40.
1990	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	6.	33.	39.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	10.	6.	1.0	0.0	0.0	0.0	10.8	10.8	0.0	172.	136.	308.

Table 7-41. (11 of 15) Individual Program Cost Breakdown - TDRS 4 Configuration C (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES PAYLOADS NEW	REFURB	LAUNCH VEHICLES T3D/C		T3C		SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	22.	8.	30.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	52.	16.	68.
1978	3.	0.	1.0	0.0	1.0	0.0	0.0	0.0	0.0	19.	6.	25.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1981	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	3.	2.	5.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	0.	6.
1983	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	5.	6.	11.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	12.	0.	12.
1985	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	9.	33.	42.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	2.	0.	2.
1987	0.	0.	0.0	0.0	0.0	0.0	.3	.3	0.0	3.	4.	7.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	12.	0.	12.
1989	1.	1.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	12.	11.	23.
1990	1.	0.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	2.	11.	13.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	7.	3.	1.0	0.0	1.0	0.0	6.0	6.0	0.0	161.	97.	258.

Table 7-41. (12 of 15) Individual Program Cost Breakdown - TDRS 4 Configuration D

LAUNCHED FROM ETR

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	LAUNCH VEHICLES T3D/C		T3D		SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	21.	7.	28.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	52.	14.	66.
1978	3.	0.	1.0	0.0	1.0	0.0	0.0	0.0	0.0	18.	6.	24.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1981	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	3.	2.	5.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	0.	6.
1983	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	5.	5.	10.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	11.	0.	11.
1985	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	8.	33.	41.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	2.	0.	2.
1987	0.	0.	0.0	0.0	0.0	0.0	.3	.3	0.0	3.	4.	7.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	12.	0.	12.
1989	1.	1.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	11.	11.	22.
1990	1.	0.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	2.	11.	13.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	7.	3.	1.0	0.0	1.0	0.0	6.0	6.0	0.0	156.	93.	249.

(The table's group headings are SPACE TRANSPORTATION SYSTEM, SCHEDULED QUANTITIES and PROGRAM DIRECT COST.)

Table 7-41. (13 of 15) Individual Program Cost Breakdown - TDRS 5 Configuration B

LAUNCHED FROM ETR

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES LAUNCH VEHICLES T3D/C				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	4.	23.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	45.	9.	54.
1978	3.	0.	1.0	0.0	0.0	0.0	0.0	0.0	0.0	17.	3.	20.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	2.	0.	2.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.	0.	4.
1981	1.	0.	0.0	0.0	0.0	0.0	.8	.8	0.0	9.	9.	18.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	0.	19.
1983	3.	0.	0.0	0.0	0.0	0.0	1.5	1.5	0.0	17.	16.	33.
1984	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	7.	34.	41.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	5.	0.	5.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	12.	0.	12.
1987	2.	0.	0.0	0.0	0.0	0.0	2.3	2.3	0.0	11.	25.	36.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	23.	0.	23.
1989	2.	4.	0.0	0.0	0.0	0.0	4.5	4.5	0.0	24.	50.	74.
1990	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	7.	33.	40.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	13.	8.	1.0	0.0	0.0	0.0	15.1	15.1	0.0	221.	183.	404.

Table 7-41. (14 of 15) Individual Program Cost Breakdown - TDRS 5 Configuration C
(Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	T3D/C		T3C		SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	PROGRAM DIRECT COST LAUNCH VEHICLES	PROGRAM DIRECT COST TOTAL
			SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES LAUNCH VEHICLES									
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	21.	8.	29.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	50.	16.	66.
1978	3.	0.	1.0	0.0	1.0	0.0	0.0	0.0	0.0	19.	6.	25.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1981	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	9.	2.	11.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	0.	19.
1983	3.	0.	0.0	0.0	0.0	0.0	1.5	1.5	0.0	11.	17.	28.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	12.	0.	12.
1985	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	10.	33.	43.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	3.	0.	3.
1987	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	7.	6.	13.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	0.	19.
1989	2.	1.	0.0	0.0	0.0	0.0	1.5	1.5	0.0	20.	16.	36.
1990	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	7.	34.	41.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	11.	5.	1.0	0.0	1.0	0.0	9.7	9.7	0.0	209.	138.	347.

Table 7-41. (15 of 15) Individual Program Cost Breakdown - TDRS 5 Configuration D (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES PAYLOADS NEW	REFURB	LAUNCH VEHICLES T3D/C		T3C		SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1974	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	21.	8.	29.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	49.	16.	65.
1978	3.	0.	1.0	0.0	1.0	0.0	0.0	0.0	0.0	18.	6.	24.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.	0.	1.
1981	0.	0.	0.0	0.0	0.0	0.0	.2	.2	0.0	9.	2.	11.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	0.	19.
1983	3.	0.	0.0	0.0	0.0	0.0	1.5	1.5	0.0	10.	17.	27.
1984	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	12.	0.	12.
1985	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	10.	33.	43.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	3.	0.	3.
1987	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	7.	6.	13.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	19.	0.	19.
1989	2.	1.	0.0	0.0	0.0	0.0	1.5	1.5	0.0	19.	16.	35.
1990	1.	2.	0.0	0.0	0.0	0.0	3.0	3.0	0.0	8.	34.	42.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	11.	5.	1.0	0.0	1.0	0.0	9.7	9.7	0.0	206.	138.	344.

The cost analysis of the TDRS program assumed that the Tug could dock and retrieve a spinning satellite. This capability quite probably would be furnished as standard equipment on the Tug and funded by either the Tug or payload programs. Estimates for the cost of this equipment are provided to show that the results of the cost analysis would not be altered by excluding these costs.

Docking mechanism concepts examined included the Apollo probe and drogue system developed by North American Rockwell, an Aerospace Corporation adaptation of the Apollo system for spinning satellites, and a McDonnell Douglas Astronautics concept. The concept considered for costing was a modification of the basic Apollo probe mechanism for active docking with a stable or spinning satellite. The active docking mechanism, which weighs 37.6 kg (83 lb), is capable of arresting rotational motion, retracting the probe, and attaching the satellite to the Tug. Incremental costs for the mechanism only are provided since the basic docking system is currently available. These costs include only the active hardware and exclude all passive portions of the docking system such as satellite support ring, fittings and latches, guide and protection arms, etc. Program RDT&E cost increments are estimated as approximately \$2.75 million based on CER estimates. Unit cost estimates are \$660,000 and are based on discussions with the vendor.

7.3.3 System Test Satellite Program Characteristics and Results

The demonstration program for test satellites consists of individual missions of 5-year duration, each requiring two satellites launched in consecutive years. Four different missions are conducted in series. Each requires new mission equipment R&D, but all use the same spacecraft. Between 1980 and 1985, three missions are initiated consecutively. Then, in 1987 another mission is initiated. The program as set requires eight satellites to perform the missions in an expendable mode. In a reusable mode it requires six new satellites and two refurbishments. The satellites are launched by the Shuttle and are placed in synchronous orbit. Upper stages used in an expendable mode are the Agena, Transtage, and Centaur. Tugs are used for the reusable mode. Intelsat IV satellite designs are used as an example of a System Test Satellite.

Program cost estimates are presented in Tables 7-42, 7-43, and 7-44 for configurations A, B, C, and D. Configuration A is the baseline expendable satellite and is launched with the Shuttle and expendable upper stage. The other configurations are identical to those used in the TDRS program. Characteristics of the individual configurations are provided in Table 7-45 with weights, costs, schedules, and quantities. Launch schedules and direct charges are provided in Table 7-46 as determined from the capture analysis. Trip sharing is used when feasible according to the mission model.

The overall results indicate that in programs of this nature, with a high ratio of new satellites and low unit cost, the expendable mode may be conducted at lower cost and less risk.

Table 7-42.

PROGRAM DIRECT COST SUMMARY
DIRECT PROGRAM COSTS
(MILLIONS OF 1971 DOLLARS)

	SPACE TRANSPORTATION SYSTEM PAYLOAD TOTAL	LNCH VEH DIRECT	PROGRAM DIRECT
SYST. TEST CONF. A B/L EXPENDABLE - 6 YR MMD			
TEST CONF. A	134.	50.	184.
SYST. TEST CONF. B B/L REUSEABLE - 6 MMD			
TEST CONF. B	128.	66.	194.
SYST. TEST CONF. C OPTI. REUSEABLE - 7 YR MMD			
TEST CONF. C	141.	70.	211.
SYST. TEST CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD			
TEST CONF. D	140.	70.	210.

Table 7-43

PROGRAM DIRECT COST SUMMARY
DIRECT PROGRAM COSTS
(MILLIONS OF 1971 DOLLARS)
SPACE TRANSPORTATION SYSTEM

	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	1992
SYST. TEST CONF. A P/L EXPENDABLE - 6 YR MMD																		
TEST CONF. A	0.	0.	0.	9.	15.	25.	24.	21.	23.	16.	13.	14.	17.	7.	0.	0.	0.	0.
SYST. TEST CONF. B P/L REUSEABLE - 6 MMD																		
TEST CONF. B	0.	0.	0.	7.	16.	25.	25.	23.	23.	17.	17.	10.	18.	13.	0.	0.	0.	0.
SYST. TEST CONF. C OPTI. REUSEABLE - 7 YR MMD																		
TEST CONF. C	0.	0.	0.	11.	20.	26.	27.	24.	24.	18.	18.	11.	20.	12.	0.	0.	0.	0.
SYST. TEST CONF. D OPTI. REUSEABLE SHORT - 7 YR MMD																		
TEST CONF. D	0.	0.	0.	10.	20.	26.	27.	24.	24.	18.	18.	11.	20.	12.	0.	0.	0.	0.

Table 7-44. TDRS Case

	PAYLOAD TYPE	PAYLOAD TOTALS PAYLOAD QUANTITIES NEW	REFURBED	PAYLOAD PROGRAM COSTS RDTE	INVEST	OPS	TOTAL
SYST. TEST CONF. A	B/L EXPENDABLE - 5 YR MMD						
TEST CONF. A	EXPENDABLE	8.	0.	53.	74.	7.	134.
SYST. TEST CONF. B	B/L REUSEABLE - 5 MMD						
TEST CONF. B	CURR REUSABLE	6.	2.	53.	61.	14.	128.
SYST. TEST CONF. C	OPTI. REUSEABLE - 7 YR MMD						
TEST CONF. C	CURR REUSABLE	6.	2.	54.	71.	16.	141.
SYST. TEST CONF. D	OPTI. REUSEABLE SHORT - 7 YR MMD						
TEST CONF. D	CURR REUSABLE	6.	2.	54.	70.	16.	140.

Table 7-45. (1 of 4) Test Configuration A

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	426	426	TYPE,	END	1.000	1.000	8.5	1.1	1.2	0.	9.	0.	9.
ELECTRICAL POWER	259	259	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	0.	10.	0.	10.
TRACKING,COMMAND	50	50	ALT,	SYNC	1.000	1.000	6.5	1.1	1.2	0.	9.	0.	9.
STABILITY,CONTROL	91	358	TYPE,	SPIN	1.000	1.000	5.7	1.2	1.3	0.	9.	0.	9.
PROPULSION	200	1530	TOT.IMP(L)0.		1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1026	2753					28.6	4.7	5.1	0.	37.	0.	37.
MISSION EQUIPMENT	350	350	COMPLXTY,	LOW	1.000	1.000	12.5	4.2	4.6	50.	34.	0.	84.
SATELLITE	1376	3103					41.1	8.9	9.7	50.	71.	0.	121.
AGE					1.000		2.9			0.	0.	0.	0.
LAUNCH SUPPORT						1.000		.8	.9	0.	0.	7.	7.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										0.	1.	0.	1.
SE AND TD										3.	2.	0.	5.
TOTAL										53.	74.	7.	134.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REVISIONS														
SPACECRAFT														0.0
MISSION EQUIPMENT		1.00		1.00		1.00			1.00					4.0
SATELLITE SCHEDULE														
NEW (EXPENDABLE)	[illegible]	1.	1.	1.	1.	1.	1.	0.	1.	1.	0.	0.	0.	8.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	[illegible]	0.	[illegible]	4.	7.	[illegible]	7.	5.	7.	7.	3.	7.	3.	0.	0.	0.	0.	53.
INVESTMENT	[illegible]	[illegible]	[illegible]	4.	[illegible]	[illegible]	[illegible]	3.	3.	7.	4.	7.	7.	2.	0.	0.	0.	74.
OPERATIONS	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	1.	1.	1.	1.	[illegible]	0.	1.	[illegible]	0.	0.	0.	7.
TOTAL	[illegible]	[illegible]	[illegible]	4.	14.	17.	17.	15.	17.	11.	7.	14.	11.	2.	0.	0.	0.	134.

Table 7-45. (2 of 4) Test Configuration B

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	510	510	TYPE,	SMP	1.000	1.000	9.7	1.4	1.6	0.	9.	0.	9.
ELECTRICAL POWER	234	254	WATTS,	573.	1.000	1.000	7.9	1.2	1.3	0.	7.	0.	7.
TRACKING,COMMAND	76	[illegible]	ALT,	SYNC	1.000	1.000	6.5	1.1	1.2	0.	7.	0.	7.
STABILITY,CONTROL	91	477	TYPE,	SPIN	1.000	1.000	5.9	1.2	1.3	0.	7.	0.	7.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1019	1365					30.0	5.1	5.5	0.	30.	0.	30.
MISSION EQUIPMENT	350	350	COMPLXTY,	LOW	1.000	1.000	12.5	4.2	4.6	50.	28.	0.	78.
SATELLITE	1369	1715					42.5	9.3	10.1	50.	58.	7.	115.
AGE					1.000		2.9			0.	0.	0.	0.
LAUNCH SUPPORT						1.000		.8	.8	0.	0.	7.	7.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										0.	1.	0.	1.
SE AND TD										3.	2.	0.	5.
TOTAL										53.	61.	14.	128.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT														0.0
MISSION EQUIPMENT		1.00		1.00		1.00			1.00					4.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	0.	1.	1.	1.	1.	1.	1.	0.	0.	0.	0.	0.	0.	6.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	0.	0.	1.	1.	0.	0.	0.	2.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	0.	0.	4.	7.	6.	7.	6.	7.	3.	3.	7.	3.	0.	0.	0.	0.	53.
INVESTMENT	0.	0.	0.	2.	8.	10.	10.	10.	10.	7.	3.	1.	0.	0.	0.	0.	0.	61.
OPERATIONS	0.	0.	0.	0.	0.	2.	1.	1.	1.	1.	0.	2.	4.	2.	0.	0.	0.	14.
TOTAL	0.	0.	0.	5.	15.	18.	18.	17.	18.	11.	6.	10.	7.	2.	0.	0.	0.	128.

Table 7-45. (3 of 4) Test Configuration C

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TOPS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	584	584	TYPE,	END	1.000	1.000	10.0	1.5	1.7	0.	9.	0.	9.
ELECTRICAL POWER	380	380	WATTS,	570.	1.000	1.000	7.9	1.2	1.3	0.	7.	0.	7.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	0.	12.	0.	12.
STABILITY,CONTROL	145	981	TYPE,	SPIN	1.000	1.000	7.5	1.8	2.0	0.	11.	0.	11.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1236	2122					33.7	6.5	7.0	0.	39.	0.	39.
MISSION EQUIPMENT	358	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	51.	29.	0.	80.
SATELLITE	1594	2490					46.4	10.8	11.7	51.	68.	8.	127.
AGE					1.000		2.9			0.	0.	0.	0.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	8.	8.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										0.	1.	0.	1.
SE AND TD										3.	2.	0.	5.
TOTAL										54.	71.	16.	141.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT														0.0
MISSION EQUIPMENT		1.00		1.00		1.00			1.00					4.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	0.	1.	1.	1.	1.	1.	1.	0.	0.	0.	0.	0.	0.	6.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	0.	0.	1.	1.	0.	0.	0.	2.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	0.	0.	5.	7.	6.	7.	6.	7.	3.	3.	7.	3.	0.	0.	0.	0.	54.
INVESTMENT	0.	0.	0.	5.	9.	11.	11.	11.	11.	8.	3.	1.	1.	0.	0.	0.	0.	71.
OPERATIONS	0.	0.	0.	0.	1.	1.	1.	1.	1.	1.	1.	3.	5.	1.	0.	0.	0.	16.
TOTAL	0.	0.	0.	10.	17.	18.	19.	18.	19.	12.	7.	11.	9.	1.	0.	0.	0.	141.

Table 7-45. (4 of 4) Test Configuration D

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

TDRS CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		COST FACTOR DEV	COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	560	980	TYPE,	SNP	1.000	1.000	9.5	1.4	1.5	0.	8.	0.	8.
ELECTRICAL POWER	380	493	WATTS,	575.	1.000	1.000	7.9	1.2	1.3	0.	7.	0.	7.
TRACKING,COMMAND	77	77	ALT,	SYNC	1.000	1.000	8.2	1.9	2.1	0.	12.	0.	12.
STABILITY,CONTROL	145	329	TYPE,	SPIN	1.000	1.000	7.4	1.8	1.9	0.	11.	0.	11.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	1182	1966					33.0	6.3	6.8	0.	38.	0.	38.
MISSION EQUIPMENT	368	368	COMPLXTY,	LOW	1.000	1.000	12.7	4.3	4.7	51.	29.	0.	80.
SATELLITE	1550	2334					45.7	10.6	11.5	51.	57.	8.	126.
AGE					1.000		2.9			0.	0.	0.	0.
LAUNCH SUPPORT						1.000		1.0	1.0	0.	0.	8.	8.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										0.	1.	0.	1.
SE AND TD										3.	2.	0.	5.
TOTAL										54.	70.	16.	140.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT														0.0
MISSION EQUIPMENT		1.00		1.00		1.00			1.00					4.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	0.	1.	1.	1.	1.	1.	1.	0.	0.	0.	0.	0.	0.	6.
REFURB (RATE=.390)	0.	0.	0.	0.	0.	0.	0.	0.	1.	1.	0.	0.	0.	2.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	0.	0.	0.	5.	7.	6.	7.	6.	7.	3.	3.	7.	3.	0.	0.	0.	0.	54.
INVESTMENT	0.	0.	0.	4.	9.	11.	11.	11.	11.	8.	3.	1.	1.	0.	0.	0.	0.	70.
OPERATIONS	0.	0.	0.	0.	1.	1.	1.	1.	1.	1.	1.	3.	5.	1.	0.	0.	0.	16.
TOTAL	0.	0.	0.	9.	17.	18.	19.	18.	19.	12.	7.	11.	9.	1.	0.	0.	0.	140.

Table 7-46. (1 of 4)

INDIVIDUAL PROGRAM COST BREAKDOWN, MILLIONS OF 1971 DOLLARS)
TEST CONF. A
LAUNCHED FROM ETR

	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES									PROGRAM DIRECT COST		
			LAUNCH VEHICLES									
FISCAL YEAR	PAYLOADS NEW	REFURB	A				SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	8.	1.	9.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	14.	1.	15.
1980	1.	0.	.5	0.0	0.0	0.0	.5	0.0	0.0	17.	8.	25.
1981	1.	0.	.5	0.0	0.0	0.0	.5	0.0	0.0	17.	7.	24.
1982	1.	0.	.5	0.0	0.0	0.0	.5	0.0	0.0	16.	5.	21.
1983	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	17.	6.	23.
1984	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	11.	5.	16.
1985	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	7.	6.	13.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	14.	0.	14.
1987	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	11.	6.	17.
1988	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	2.	5.	7.
1989	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	8.	0.	1.5	0.0	0.0	0.0	4.0	2.5	0.0	134.	50.	184.

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS (OF 1971 DOLLARS)
TEST CONF. [illegible]
LAUNCHED FROM ETR

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	LAUNCH VEHICLES T				SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	1.	7.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	15.	1.	16.
1980	1.	0.	0.0	.5	0.0	0.0	.5	0.0	0.0	18.	7.	25.
1981	1.	0.	0.0	.5	0.0	0.0	.5	0.0	0.0	18.	7.	25.
1982	1.	0.	0.0	.5	0.0	0.0	.5	0.0	0.0	17.	6.	23.
1983	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	18.	5.	23.
1984	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	11.	6.	17.
1985	1.	0.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	6.	11.	17.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	10.	0.	10.
1987	0.	1.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	7.	11.	18.
1988	0.	1.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	2.	11.	13.
1989	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	6.	2.	0.0	1.5	0.0	0.0	5.5	4.0	0.0	128.	66.	194.

Table 7-46. (3 of 4)

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
TEST CONF. C
LAUNCHED FROM ETR

FISCAL YEAR	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES PAYLOADS NEW	REFURB	LAUNCH VEHICLES C				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	10.	1.	11.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	17.	3.	20.
1980	1.	0.	0.0	0.0	.5	0.0	.5	0.0	0.0	18.	8.	26.
1981	1.	0.	0.0	0.0	.5	0.0	.5	0.0	0.0	19.	8.	27.
1982	1.	0.	0.0	0.0	.5	0.0	.5	0.0	0.0	18.	6.	24.
1983	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	19.	5.	24.
1984	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	12.	6.	18.
1985	1.	0.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	7.	11.	18.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	11.	0.	11.
1987	0.	1.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	9.	11.	20.
1988	0.	1.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	1.	11.	12.
1989	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	6.	2.	0.0	0.0	1.5	0.6	5.5	4.0	0.0	141.	70.	211.

Table 7-46. (4 of 4)

INDIVIDUAL PROGRAM COST BREAKDOWN MILLIONS OF 1971 DOLLARS)
TEST CONF, D.
LAUNCHED FROM ETR

	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES		LAUNCH VEHICLES							PROGRAM DIRECT COST		
FISCAL YEAR	PAYLOADS NEW	REFURB			C		SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	9.	1.	10.
1979	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	17.	3.	20.
1980	1.	0.	0.0	0.0	.5	0.0	.5	0.0	0.0	18.	8.	26.
1981	1.	0.	0.0	0.0	.5	0.0	.5	0.0	0.0	19.	8.	27.
1982	1.	0.	0.0	0.0	.5	0.0	.5	0.0	0.0	18.	6.	24.
1983	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	19.	5.	24.
1984	1.	0.	0.0	0.0	0.0	0.0	.5	.5	0.0	12.	6.	18.
1985	1.	0.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	7.	11.	18.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	11.	0.	11.
1987	0.	1.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	9.	11.	20.
1988	0.	1.	0.0	0.0	0.0	0.0	1.0	1.0	0.0	1.	11.	12.
1989	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	5.	2.	0.0	0.0	1.5	0.0	5.5	4.0	0.0	140.	70.	210.

7.4 OBSERVATORY - HEAO PROGRAM COST ANALYSIS

7.4.1 Summary

The basic HEAO program consists of a HEAO-C payload providing continuous coverage between 1979 and 1983, and various follow-on spacecraft with redesigned mission equipment in use for five-year periods beginning in 1983.

The results of the observatory cost analysis of the HEAO program consist of three tradeoffs involving:

(1) A comparison of candidate man-tended and automated spacecraft concepts

(2) A comparison of satellite design life

(3) The impact of accessibility for yearly adjustments

(4) A comparison of the revisits made with the ground refurbishment mode

The criterion recommended for selection of the spacecraft approach is lowest total program cost for the HEAO-C (1975-83) program, with consideration of the impact on the total program cost (1975-1990). The results indicate that a scientific program with developed spacecraft hardware (i.e., HEAO-C uses HEAO-A and HEAO-B hardware) and frequent experiment changes should use a low initial cost, minimum design modification, and reusable spacecraft approach. The alternative would be a minimum total cost program with major initial spacecraft design changes to develop an on-orbit serviceable (revisitable) satellite. Low initial program cost is an important evaluation criterion since it permits the most flexible long-term program with the least amount of initial risk.

The comparisons provided are based on relative cost estimates of payloads using current baseline estimating procedures. If absolute cost estimates are to be made, further analysis of large/low cost payload and hardware inheritance effects will be required.

7.4.1.1 Candidate Concepts Comparison

Direct program cumulative costs over various periods of the program are shown in Table 7-47 for the three basic spacecraft configuration concepts. Case A represents the costs for either configuration concept A or B, which represent the baseline with minimum modifications. Case B represents the cost for a modularized payload with on-orbit docking capability as provided by spacecraft configuration concepts C and E. Case C represents the cost for a man-tended spacecraft with IVA capability. All payloads in this comparison have 2-yr design life. Case A has the lowest program costs over the initial operating years through 1981. These costs are indicative of the HEAO-C operating period. The minimum total program cost, however, is associated with Case B or Case C, as shown by the net present value for infinite horizon and 10-percent discount rate. The factor which causes the minimum cost to switch from Case A to B or C is the requirement for accessibility to the payload once per year in order to make adjustments. This is accomplished on the ground in Case A and by on-orbit revisits in Cases B and C. If the requirement is reduced to access once every two years, as required for refurbishment, then the minimum total cost program would be Case A. The impact of the annual payload access requirement in Case A is discussed in more detail in the following paragraphs.

Results of this tradeoff, and that for accessibility, lead to the recommendation that Case A be selected for the HEAO program. The basis for the recommendation is minimum cost in the early years of the program, which lowers the risk and retains some flexibility for changes in the later years.

Table 7-47. HEAO Program Direct Cost Estimates — Candidate Concepts Comparison,[1] 2-yr Design Life

(Millions of 1971 Dollars)

Operating Period	Direct Program Cumulative Totals			
	1979	1979 - 1981	1979 - 1990	Net Present Value[2]
Cases				
A. HEAO-C, Mod A/B	225	305	784	363
B. HEAO-C, Mod C/E	283	355	701	329
C. HEAO-C, Mod D	293	368	724	339

(1) Payload access for adjustments once each year by retrieval (Case A) or On-Orbit Visit (Cases B, C)

(2) Infinite Horizon, 10-percent Discount

The final selection of a program should involve other tradeoffs with truncated phases of the program and less than continuous observations coverage by the satellites.

7.4.1.2 Design Life Comparison

The influence of design life on program costs was examined for its effect on spacecraft configuration concepts A and B. Configurations A and B are refurbished on the ground. Access for experiment adjustments is restricted to the refurbishment interval, once each satellite lifetime. Cost estimates shown in Table 7-48 indicate that the minimum initial cost is associated with Case 2A for a 2-yr life. In terms of total program cost, as shown by the net present value, the most dramatic reduction in cost is achieved in going from 1-yr to 2-yr design life. The lowest total cost is achieved by going to the longest life as shown for Case 4A with 5-yr life. Cases 3A and 4A are equivalent in total cost because they are within the relative tolerance of the estimates.

A determination of the best design life for HEAO-C must consider the configuration characteristics, initial and final program costs, access requirements by the scientists, and on-orbit observational coverage time. A requirement for accessibility to the payload at least once every two years would clearly lead to the selection of Case 2A as the best configuration since cases 3A and 4A would require more funds than those shown. The increase in cost involves more Shuttle flights and possibly more payloads if continuous coverage is required. Without the requirement for accessibility Case 3A with a 4-yr design life appears to be the selection based on HEAO-C and HEAO total program cost values.

7.4.1.3 Impact of Annual Accessibility

Access to the payload in order to make adjustments to the mission equipment and the spacecraft is a requirement desired by the scientist. The cost impact owing to the frequency with which these adjustments can be

Table 7-48. HEAO Program Direct Cost Estimates — Design Life Comparison HEAO-C, Configuration A/B

(Millions of 1971 Dollars)

Operating Period	Direct Program Cumulative Totals			
	1979	1979 - 1981	1979 - 1990	Net Present Value(1)
Cases				
1A. 1-yr life	204	304	839	387
2A. 2-yr life	174	271	637	301
3A. 4-yr life	196	251	600	278
4A. 5-yr life	211	235	585	264

(1) infinite horizon, 10-percent discount

made was provided in the data for configuration A/B of the previous comparisons of concepts and design life. Access to the payload each year is provided in the cost of Case A (Table 7-47), and access once every two years is provided in the cost of Case 2A (Table 7-48). The difference between the two cases is shown in Table 7-49. The impact represents more than just the cost of extra Shuttle flights. In order to maintain continuous coverage during ground based refurbishment an additional satellite must be purchased to replace the one being retrieved, refurbished, and adjusted on the ground. The costs associated with the additional satellite, additional refurbishment, and the additional transportation costs amount to approximately $150 million over the total program. The requirement for increasing the satellite accessibility for experiment revisit from once every two years to annually therefore increases the total program cost by 23 percent and the initial cost by nearly 30 percent. Because of the high cost impact, the accessibility requirement for configurations A and B should be reduced to operate HEAO-C with access frequency similar to HEAO-A and HEAO-B.

For cases (B and C of Table 7-47) where payload adjustments are conducted during on-orbit maintenance revisits the cost of yearly access is much less than that for Case A. The cost difference involved in achieving yearly revisits rather than every other year is approximately $30 million in Shuttle transportation costs.

7.4.2 Candidate Concepts Comparison

7.4.2.1 Program Characteristics

The HEAO program is a follow-on to the basic HEAO-A and HEAO-B programs to be flown in 1975 and 1977. The program will consist of HEAO-C operating from 1979 to 1983; thereafter two new HEAOs will be launched in 1983 and 1984. The new HEAOs consist of new mission equipment but

Table 7-49. HEAO Program Direct Cost Estimates — Impact of Accessibility for Yearly Adjustments HEAO-C, Configuration A/B, 2-yr Life

(Millions of 1971 Dollars)

	Direct Program Cumulative Totals			
Operating Period	1979	1979 - 1981	1979 - 1990	Net Present Value(1)
Cases				
A. 2-yr life, access once per year	225	305	784	363
2A. 2-yr life, access once every two years	174	271	637	301
Difference	51	34	147	62

(1) infinite horizon, 10-percent discount

the same spacecraft as HEAO-C.

The purpose of this section of the study is to determine the best configuration concept for the HEAO-C within the context of the total program and involving all the HEAO satellites from 1979 to 1990. The assumption made for this study is that HEAO-A and HEAO-B spacecraft hardware and technology are available for the HEAO-C programs. With this baseline, three configuration concepts were selected for HEAO-C analysis.

The first case (Case A) represents configuration A or B, which are minimum modification versions of the baseline to permit launching from the Shuttle. These satellites must be retrieved and refurbished on the ground. Adjustments to the mission equipment also require retrieval and changes made on the ground. Because this configuration concept is similar to the baseline, an inheritance factor of 75 percent on development is used for the spacecraft except structures. That is, the development cost of all subsystems is considered to be 25 percent of a new subsystem except for structures, which is 100 percent because it is a new design. Mission equipment is also considered to be 100 percent of a new design.

The second case (Case B) represents a new HEAO-C design (configuration C or E) with modular subsystems, docking system, and automation to permit on-orbit maintenance by the use of manipulators. The third case (Case C) represents configuration D, which is a version of configuration C with IVA capability for man-tending of the payload. In Cases B and C no inheritance of HEAO-A or HEAO-B hardware was assumed since most of the development work must be redone, with new packaging, new test articles, and a new test program. All satellites compared here had design lives of two years.

Program requirements include continuous on-orbit coverage by the satellites and yearly accessibility for mission equipment adjustments. Case A requires retrieval and deployment yearly between 1979 and 1990 to accomplish refurbishments and/or adjustments on the ground. Refurbishment factors were considered to be 32 percent of the unit cost. In Cases B and C the satellite is maintained on-orbit with yearly visits. A maintenance factor of 10 percent each year, or 20 percent each MMD, was used. Beginning in 1984 the mission equipment is to be changed for the HEAO program every 5 years by a ground refurbishment costed at 32 percent of the unit cost. Cost estimates were generated for each case on the basis of the above considerations and a capture analysis to establish the launch schedules and trip charges.

7.4.2.2 Results

The program direct cost summaries are presented in Table 7-50 with the total cost during the period 1975-1990, and in Table 7-51 with the associated cost streams. A breakdown of the payload program phase costs are presented in Table 7-52 along with the quantities of new and refurbished payloads used. Payload weights, costs, and schedules for each are displayed in Table 7-53. The Shuttle launch schedule, with the trip charge for full or shared rides established by the capture analysis, is presented in Table 7-54.

7.4.3 Design Life Comparison

7.4.3.1 Program Characteristics

The effect of design life on the cost of the HEAO program for configuration A/B was examined in this study. Satellite lifetimes from one to five years were considered. Program requirements for this particular study were slightly different from those described in section 7.4.2 concerning concept comparisons, in that the adjustments to the mission equipment were limited to the refurbishment schedule of once each satellite lifetime. With regard to scheduling launches and buying new satellites the capture analysis was based on continuous on-orbit coverage with ground refurbishment. In order to ensure minimum costs, several programs were addressed in the capture

Table 7-50. Program Direct Cost Summary - Direct Program Costs, Space Transportation System (Millions of 1971 Dollars)

	PAYLOAD TOTAL	LNCH VEH DIRECT	PROGRAM DIRECT
CASE A (HEAO-C, CONFIG. A/B 1979 THRU 1990) HEAO-C MOD A/B	695.	89.	784.
CASE B (HEAO-C, CONFIG. C/E 1979 THRU 1990) HEAO-C MOD C/E	624.	77.	701.
CASE C (HEAO-C, CONFIG. D 1979 THRU 1990) HEAO-C MOD D	647.	77.	724.

Table 7-51. **Program Direct Cost Summary - Direct Program Costs - Space Transportation System (Millions of 1971 Dollars)**

	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	1992
CASE A (HEAO-C, CONFIG. A/B 1979 THRU 1990)																		
HEAO-C MOD A/B	8.	24.	46.	74.	73.	37.	43.	78.	109.	91.	46.	43.	47.	40.	20.	5.	0.	0.
CASE B (HEAO-C, CONFIG. C/E 1979 THRU 1990)																		
HEAO-C MOD C/E	22.	55.	84.	80.	42.	24.	48.	74.	66.	42.	24.	26.	29.	44.	34.	7.	0.	0.
CASE C (HEAO-C, CONFIG. D 1979 THRU 1990)																		
HEAO-C MOD D	23.	57.	87.	83.	43.	25.	50.	76.	68.	43.	24.	26.	30.	46.	35.	7.	0.	0.

Table 7-52. HEAO Case

	PAYLOAD TOTALS						
	PAYLOAD QUANTITIES			PAYLOAD PROGRAM COSTS			
	TYPE	NEW	REFURBED	RDTE	INVEST	OPS	TOTAL
CASE A (HEAO-C, CONFIG. A/B 1979 THRU 1990)							
HEAO-C MOD A/B	CURR REUSABLE	4.	7.	171.	317.	207.	695.
CASE B (HEAO-C, CONFIG. C/E 1979 THRU 1990)							
HEAO-C MOD C/E	CURR REUSABLE	2.	2.	275.	191.	158.	624.
CASE C (HEAO-C, CONFIG. D 1979 THRU 1990)							
HEAO-C MOD D	CURR REUSABLE	2.	2.	281.	199.	167.	647.

Table 7-53. (1 of 3) HEAO-C Mod A/B

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

HEAO CASE SUBSYSTEM	WEIGHTS OPY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	7304	7304	TYPE,	EXD	1.000	1.000	18.3	8.4	8.4	18.	33.	0.	51.
ELECTRICAL POWER	981	981	WATTS,	820.	1.000	1.000	1.9	1.6	1.6	2.	6.	0.	8.
TRACKING,COMMAND	178	178	ALT,	LOW ORBIT	1.000	1.000	2.4	5.2	5.2	2.	20.	0.	22.
STABILITY,CONTROL	1079	1927	TYPE,	3-AXIS	1.000	1.000	19.2	30.6	30.6	19.	119.	0.	135.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	9542	10390					41.8	45.8	45.8	41.	178.	0.	219.
MISSION EQUIPMENT	7175	7175	COMPLXTY,	LOW	1.000	1.000	28.3	24.7	24.7	113.	130.	0.	243.
SATELLITE	16717	17565					70.1	70.5	70.5	154.	308.	158.	620.
AGE					1.000		5.9			6.	0.	0.	6.
LAUNCH SUPPORT						1.000		4.4	4.4	0.	0.	48.	48.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	5.	0.	6.
SE AND TD										10.	4.	1.	15.
TOTAL										171.	317.	207.	695.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00				1.00	1.00					1.00			4.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	1.	1.	0.	0.	0.	1.	1.	0.	0.	0.	0.	0.	0.	4.
REFURB (RATE=.320)	0.	0.	0.	1.	1.	1.	0.	1.	1.	1.	1.	0.	0.	7.

FISCAL YEAR FUNDING	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
RDTE	8.	24.	28.	14.	9.	12.	20.	16.	8.	2.	3.	9.	11.	5.	2.	0.	0.	171.
INVESTMENT	0.	0.	18.	58.	55.	15.	4.	30.	67.	56.	14.	0.	0.	0.	0.	0.	0.	317.
OPERATIONS	0.	0.	0.	2.	4.	2.	14.	27.	29.	18.	16.	27.	27.	27.	14.	0.	0.	207.
TOTAL	8.	24.	46.	74.	68.	29.	38.	73.	104.	76.	33.	36.	38.	32.	16.	0.	0.	695.

Table 7-53. (2 of 3) HEAO-C Mod C/E

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

HEAO CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	8628	8628	TYPE,	EXO	1.000	1.000	18.7	9.0	9.0	19.	18.	0.	37.
ELECTRICAL POWER	981	981	WATTS,	920.	1.000	1.000	7.5	1.6	1.6	7.	3.	0.	10.
TRACKING,COMMAND	178	178	ALT,	LOW ORBIT	1.000	1.000	9.5	5.2	5.2	10.	10.	0.	20.
STABILITY,CONTROL	1128	2177	TYPE,	3-AXIS	1.000	1.000	85.5	34.6	34.6	86.	69.	0.	155.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	10915	11964					121.3	50.4	50.4	122.	100.	0.	222.
MISSION EQUIPMENT	7175	7175	COMPLXTY,	LOW	1.000	1.000	28.3	24.7	24.7	113.	83.	0.	196.
SATELLITE	18090	19139					149.6	75.1	75.1	235.	183.	138.	556.
AGE					1.000		23.5			23.	0.	0.	23.
LAUNCH SUPPORT						1.000		4.7	4.7	0.	0.	19.	19.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	3.	0.	4.
SE AND TD										16.	5.	1.	22.
TOTAL										275.	191.	158.	624.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00				1.00	1.00					1.00			4.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	1.	0.	0.	0.	1.	0.	0.	0.	0.	0.	0.	0.	0.	2.
REFURB (RATE=.320)	0.	0.	0.	0.	0.	1.	0.	0.	0.	0.	1.	0.	0.	2.
MAINTENANCE FLTS	0.000	.100	.100	.100	0.000	.100	.200	.200	.200	.100	0.000	.100	0.000	1.200

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	22.	55.	64.	33.	16.	12.	19.	16.	7.	2.	3.	9.	10.	5.	2.	0.	0.	275.
INVESTMENT	0.	0.	20.	44.	15.	0.	20.	48.	25.	3.	0.	0.	4.	9.	2.	0.	0.	191.
OPERATIONS	0.	0.	0.	3.	5.	4.	4.	5.	29.	31.	9.	9.	6.	26.	26.	1.	0.	158.
TOTAL	22.	55.	84.	80.	37.	16.	43.	69.	61.	36.	12.	18.	20.	40.	30.	1.	0.	624.

Table 7-53. (3 of 3) HEAO-C Mod D

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

HEAO CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	11107	11107	TYPE,	EXP	1.000	1.000	19.3	10.2	10.2	19.	20.	0.	39.
ELECTRICAL POWER	981	981	WATTS,	320.	1.000	1.000	7.5	1.6	1.6	7.	3.	0.	10.
TRACKING,COMMAND	178	178	ALT,	LOW ORBIT	1.000	1.000	9.5	5.2	5.2	10.	10.	0.	20.
STABILITY,CONTROL	1164	2365	TYPE,	3-AXIS	1.000	1.000	91.9	37.5	37.5	92.	75.	0.	167.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	13430	14626					128.2	54.5	54.5	128.	108.	0.	236.
MISSION EQUIPMENT	7175	7175	COMPLXTY,	LOW	1.000	1.000	28.3	24.7	24.7	113.	83.	0.	196.
SATELLITE	20605	21801					156.5	79.2	79.2	241.	191.	146.	578.
AGE					1.000		23.5			23.	0.	0.	23.
LAUNCH SUPPORT						1.000		4.9	4.9	0.	0.	20.	20.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	3.	0.	4.
SE AND TD										16.	5.	1.	22.
TOTAL										281.	199.	167.	647.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00				1.00	1.00					1.00			4.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	1.	0.	0.	0.	1.	0.	0.	0.	0.	0.	0.	0.	0.	2.
REFURB (RATE=.320)	0.	0.	0.	0.	0.	1.	0.	0.	0.	0.	1.	0.	0.	2.
MAINTENANCE FLTS	0.000	.100	.100	.100	0.000	.100	.200	.200	.200	.100	0.000	.100	0.0000	1.200

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	23.	57.	66.	34.	16.	12.	19.	16.	7.	2.	3.	9.	10.	5.	2.	0.	0.	281.
INVESTMENT	0.	0.	21.	46.	17.	0.	21.	50.	26.	3.	0.	0.	4.	9.	2.	0.	0.	199.
OPERATIONS	0.	0.	0.	3.	5.	5.	5.	5.	30.	32.	9.	9.	7.	28.	28.	1.	0.	167.
TOTAL	23.	57.	87.	83.	38.	17.	45.	71.	63.	37.	12.	18.	21.	42.	32.	1.	0.	647.

Table 7-54. Individual Program Cost Breakdown (1 of 3) - HEAO-C Mod A/B (Millions of 1971 Dollars)

LAUNCHED FROM ETR

	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES										PROGRAM DIRECT COST		
			LAUNCH VEHICLES										
FISCAL YEAR	PAYLOADS NEW	REFURB					SHTL SHTL	TUG	TUG EXP		PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		8.	0.	8.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		24.	0.	24.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		45.	0.	46.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		74.	0.	74.
1979	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0		68.	5.	73.
1980	1.	0.	0.0	0.0	0.0	0.0	.7	0.0	0.0		29.	8.	37.
1981	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0		38.	5.	43.
1982	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0		73.	5.	78.
1983	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0		134.	5.	139.
1984	1.	1.	0.0	0.0	0.0	0.0	1.4	0.0	0.0		76.	15.	91.
1985	1.	0.	0.0	0.0	0.0	0.0	1.2	0.0	0.0		33.	13.	46.
1986	0.	1.	0.0	0.0	0.0	0.0	.7	0.0	0.0		36.	7.	43.
1987	0.	1.	0.0	0.0	0.0	0.0	.9	0.0	0.0		38.	9.	47.
1988	0.	1.	0.0	0.0	0.0	0.0	.7	0.0	0.0		32.	8.	40.
1989	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0		16.	4.	20.
1990	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0		0.	5.	5.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0		0.	0.	0.
TOTAL	4.	7.	0.0	0.0	0.0	0.0	8.5	0.0	0.0		695.	89.	784.

Table 7-54. Individual Program Cost Breakdown (2 of 3) - HEAO-C Mod C/E (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES: PAYLOADS NEW	PAYLOADS REFURB	LAUNCH VEHICLES				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST: PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	22.	0.	22.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	55.	0.	55.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	84.	0.	84.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	90.	0.	90.
1979	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	37.	5.	42.
1980	0.	0.	0.0	0.0	0.0	0.0	.7	0.0	0.0	15.	8.	24.
1981	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	43.	5.	48.
1982	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	69.	5.	74.
1983	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	61.	5.	66.
1984	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	35.	6.	42.
1985	0.	0.	0.0	0.0	0.0	0.0	1.2	0.0	0.0	12.	12.	24.
1986	0.	0.	0.0	0.0	0.0	0.0	.7	0.0	0.0	18.	8.	26.
1987	0.	0.	0.0	0.0	0.0	0.0	.9	0.0	0.0	20.	9.	29.
1988	0.	0.	0.0	0.0	0.0	0.0	.4	0.0	0.0	40.	4.	44.
1989	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0	30.	4.	34.
1990	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	1.	6.	7.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	2.	2.	0.0	0.0	0.0	0.0	7.3	0.0	0.0	624.	77.	701.

Table 7-54. Individual Program Cost Breakdown (3 of 3) - HEAO-C Mod D (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	PAYLOADS NEW	PAYLOADS REFURB	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES LAUNCH VEHICLES				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	23.	0.	23.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	57.	0.	57.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	87.	0.	87.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	83.	0.	83.
1979	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	38.	5.	43.
1980	0.	0.	0.0	0.0	0.0	0.0	.7	0.0	0.0	17.	8.	25.
1981	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	45.	5.	50.
1982	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	71.	5.	76.
1983	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	63.	5.	68.
1984	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	37.	6.	43.
1985	0.	0.	0.0	0.0	0.0	0.0	1.2	0.0	0.0	12.	12.	24.
1986	0.	0.	0.0	0.0	0.0	0.0	.7	0.0	0.0	18.	8.	26.
1987	0.	0.	0.0	0.0	0.0	0.0	.9	0.0	0.0	21.	9.	30.
1988	0.	0.	0.0	0.0	0.0	0.0	.4	0.0	0.0	42.	4.	46.
1989	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0	32.	4.	36.
1990	0.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	1.	6.	7.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	2.	2.	0.0	0.0	0.0	0.0	7.3	0.0	0.0	647.	77.	724.

analysis. Those requiring the lowest number of spacecraft and new mission equipment were picked. Sharing of trip charges was included when possible. The cost analysis used the basic design weight provided in order to reflect the cost impact of design life, and did not use redundancy factors. Losses for launch vehicle failures are not included in the cost estimates.

7.4.3.2 Results

The cost estimates obtained from the payload program cost model are presented in the following tables. The direct program cost summaries are shown in Tables 7-55 and 7-56. A breakdown of the payload program estimates is shown in Table 7-57, along with the quantities of new and refurbished payloads indicated. Characteristics of each payload program such as weight and cost schedules are provided in Table 7-58. The launch schedules and shared trip charges which were established in the capture analysis are presented in Table 7-59.

Table 7-55. Program Direct Cost Summary - Direct Program Costs - Space Transportation System (Millions of 1971 Dollars)

	PAYLOAD TOTAL	LNCH VEH DIRECT	PROGRAM DIRECT
CASE 1A (HEAO-C, CONFIG. A/3, 1979 THRU 1990) ONE YEAR LIFE			
HEAO, 1YR LIFE	751.	88.	839.
CASE 2A (HEAO-C, CONFIG. A/3 1979 THRU 1990) TWO YEAR LIFE			
HEAO, 2YR LIFE	593.	44.	637.
CASE 3A (HEAO-C, CONFIG. A/3 1979 THRU 1990)			
HEAO, 4YR LIFE	577.	23.	600.
CASE 4A (HEAO-C, CONFIG. A/3 1979 THRU 1990)			
HEAO, 5YR LIFE	561.	24.	585.

Table 7-56. Program Direct Cost Summary - Direct Program Costs - Space Transportation System (Millions of 1971 Dollars)

	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	1992
CASE 1A (HEAO-C, CONFIG. A/3, 1979 THRU 1990) ONE YEAR LIFE																		
HEAO, 1YR LIFE	8.	23.	43.	66.	64.	47.	53.	74.	102.	96.	65.	63.	55.	37.	28.	15.	0.	0.
CASE 2A (HEAO-C, CONFIG. A/3 1979 THRU 1990) TWO YEAR LIFE																		
HEAO, 2YR LIFE	8.	23.	44.	54.	45.	51.	46.	41.	67.	63.	43.	41.	45.	46.	20.	0.	0.	0.
CASE 3A (HEAO-C, CONFIG. A/3 1979 THRU 1990)																		
HEAO, 4YR LIFE	9.	27.	56.	69.	35.	12.	43.	92.	89.	30.	14.	43.	57.	24.	0.	0.	0.	0.
CASE 4A (HEAO-C, CONFIG. A/3 1979 THRU 1990)																		
HEAO, 5YR LIFE	10.	29.	61.	76.	35.	6.	18.	71.	128.	65.	3.	9.	15.	32.	27.	0.	0.	0.

Table 7-57. HEAO Case

	PAYLOAD TOTALS						
	PAYLOAD QUANTITIES			PAYLOAD PROGRAM COSTS			
	TYPE	NEW	REFURBED	RDTE	INVEST	OPS	TOTAL
CASE 1A (HEAO-C, CONFIG. A/B, 1979 THRU 1990) ONE YEAR LIFE							
HEAO, 1YR LIFE	CURR REUSABLE	4.	12.	166.	286.	299.	751.
CASE 2A (HEAO-C, CONFIG. A/B 1979 THRU 1990) TWO YEAR LIFE							
HEAO, 2YR LIFE	CURR REUSABLE	3.	6.	167.	264.	162.	593.
CASE 3A (HEAO-C, CONFIG. A/B 1979 THRU 1990)							
HEAO, 4YR LIFE	CURR REUSABLE	3.	2.	181.	310.	86.	577.
CASE 4A (HEAO-C, CONFIG. A/B 1979 THRU 1990)							
HEAO, 5YR LIFE	CURR REUSABLE	3.	1.	188.	319.	54.	561.

Table 7-58. (1 of 4) HEAO, 1-yr Life

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

HEAO CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	7229	7229	TYPE,	EXO	1.000	1.000	18.3	8.3	8.3	18.	33.	0.	51.
ELECTRICAL POWER	981	981	WATTS,	820.	1.000	1.000	1.9	1.6	1.6	2.	6.	0.	8.
TRACKING,COMMAND	140	140	ALT,	LOW ORBIT	1.000	1.000	2.0	4.0	4.0	2.	16.	0.	18.
STABILITY,CONTROL	986	1410	TYPE,	3-AXIS	1.000	1.000	14.6	22.4	22.4	15.	90.	0.	105.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	9336	9760					36.9	36.4	36.4	37.	145.	0.	182.
MISSION EQUIPMENT	7175	7175	COMPLXTY,	LOW	1.000	1.000	28.3	24.7	24.7	113.	133.	0.	246.
SATELLITE	16511	16935					65.2	61.1	61.1	150.	278.	235.	663.
AGE					1.000		5.9			6.	0.	0.	6.
LAUNCH SUPPORT						1.000		3.9	3.9	0.	0.	63.	63.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	4.	0.	5.
SE AND TD										9.	4.	1.	14.
TOTAL										166.	286.	299.	751.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00				1.00	1.00					1.00			4.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	1.	1.	0.	0.	0.	1.	1.	0.	0.	0.	0.	0.	0.	4.
REFURB (RATE=.320)	0.	0.	1.	1.	1.	1.	1.	2.	2.	1.	1.	1.	0.	12.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	8.	23.	27.	14.	8.	12.	20.	16.	8.	2.	3.	9.	11.	5.	0.	0.	0.	166.
INVESTMENT	0.	0.	16.	50.	47.	13.	4.	29.	63.	51.	13.	0.	0.	0.	0.	0.	0.	286.
OPERATIONS	0.	0.	0.	2.	4.	14.	24.	24.	26.	28.	37.	47.	35.	24.	24.	10.	0.	299.
TOTAL	8.	23.	43.	66.	59.	39.	48.	69.	97.	81.	53.	56.	46.	29.	24.	10.	0.	751.

Table 7-58. (2 of 4) HEAO, 2-yr Life

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

HEAO CASE SUBSYSTEM	WEIGHTS DRY	 TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	 PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	 INVEST	 OPS	 TOTAL
STRUCTURE	7354	7354	TYPE,	EXP	1.000	1.000	18.4	8.4	8.4	18.	27.	0.	45.
ELECTRICAL POWER	981	981	WATTS,	820.	1.000	1.000	1.3	1.6	1.6	2.	5.	0.	7.
TRACKING,COMMAND	178	178	ALT,	LOW ORBIT	1.000	1.000	2.4	5.2	5.2	2.	17.	0.	19.
STABILITY,CONTROL	679	1527	TYPE,	3-AXIS	1.000	1.000	15.7	24.3	24.3	16.	78.	0.	94.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	9192	10040					38.3	39.5	39.5	38.	127.	0.	165.
MISSION EQUIPMENT	7175	7175	COMPLXTY,	LOW	1.000	1.000	28.3	24.7	24.7	113.	129.	0.	242.
SATELLITE	16367	17215					66.7	64.2	64.2	151.	256.	123.	530.
AGE					1.000		5.9			6.	0.	0.	6.
LAUNCH SUPPORT						1.000		4.1	4.1	0.	0.	38.	38.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	4.	0.	5.
SE AND TD										9.	4.	1.	14.
TOTAL										167.	264.	162.	593.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00				1.00	1.00					1.00			4.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	1.	0.	1.	0.	0.	0.	1.	0.	0.	0.	0.	0.	0.	3.
REFURB (RATE=.320)	0.	0.	0.	0.	1.	1.	0.	1.	1.	1.	1.	0.	0.	6.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	8.	23.	27.	14.	8.	12.	20.	16.	8.	2.	3.	9.	11.	5.	1.	0.	0.	167.
INVESTMENT	0.	0.	17.	37.	30.	37.	18.	13.	29.	43.	21.	3.	4.	9.	3.	0.	0.	264.
OPERATIONS	0.	0.	0.	3.	2.	2.	2.	12.	25.	14.	15.	25.	25.	25.	12.	0.	0.	162.
TOTAL	8.	23.	44.	54.	40.	51.	40.	41.	62.	59.	39.	37.	40.	39.	16.	0.	0.	593.

Table 7-58. (3 of 4) HEAO, 4-yr Life

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

HEAO CASE

SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	7703	7737	TYPE,	EXP	1.000	1.000	18.5	8.6	8.6	18.	27.	0.	45.
ELECTRICAL POWER	1107	1007	WATTS,	820.	1.000	1.000	1.9	1.6	1.6	2.	5.	0.	7.
TRACKING,COMMAND	231	271	ALT,	LOW ORBIT	1.000	1.000	2.9	6.9	6.9	3.	22.	0.	25.
STABILITY,CONTROL	1262	2958	TYPE,	3-AXIS	1.000	1.000	28.2	46.9	46.9	28.	150.	0.	178.
PROPULSION	0	0	TOT.IMP,	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	10203	11899					51.4	64.0	64.0	51.	204.	0.	255.
MISSION EQUIPMENT	7175	7175	COMPLXTY,	LOW	1.000	1.000	28.3	24.7	24.7	113.	96.	0.	209.
SATELLITE	17378	19074					79.7	88.7	88.7	164.	300.	57.	521.
AGE					1.000		5.9			6.	0.	0.	6.
LAUNCH SUPPORT						1.000		5.4	5.4	0.	0.	28.	28.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	5.	0.	6.
SE AND TD										10.	5.	1.	16.
TOTAL										181.	310.	86.	577.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00				1.00	1.00				1.00				4.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	1.	0.	0.	0.	1.	1.	0.	0.	0.	0.	0.	0.	0.	3.
REFURB (RATE=.320)	0.	0.	0.	0.	0.	0.	0.	0.	1.	1.	0.	0.	0.	2.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	9.	27.	32.	16.	9.	12.	20.	16.	8.	5.	9.	11.	5.	2.	0.	0.	0.	181.
INVESTMENT	0.	0.	24.	50.	18.	0.	23.	73.	69.	18.	5.	14.	13.	3.	0.	0.	0.	310.
OPERATIONS	0.	0.	0.	3.	3.	0.	0.	3.	6.	3.	0.	18.	35.	15.	0.	0.	0.	86.
TOTAL	9.	27.	56.	69.	30.	12.	43.	92.	83.	26.	14.	43.	53.	20.	0.	0.	0.	577.

Table 7-58. (4 of 4) HEAO, 5-yr Life

PAYLOAD PROGRAM COST (MILLIONS OF 1971 DOLLARS)

HEAO CASE SUBSYSTEM	WEIGHTS DRY	WEIGHTS TOTAL	OTHER INPUTS		REDUNDANT COST FACTOR DEV	REDUNDANT COST FACTOR PROD	BASIC RDTE	AVG UNIT	FIRST UNIT	PAYLOAD PROGRAM COST ESTIMATE RDTE	INVEST	OPS	TOTAL
STRUCTURE	7880	7880	TYPE,	EXP	1.000	1.000	18.5	8.7	8.7	19.	26.	0.	45.
ELECTRICAL POWER	1[illegible]07	1[illegible]7	WATTS,	820.	1.000	1.000	1.9	1.6	1.6	2.	5.	0.	7.
TRACKING,COMMAND	236	[illegible]36	ALT,	LOW ORBIT	1.000	1.000	2.8	7.0	7.0	3.	21.	0.	24.
STABILITY,CONTROL	1740	3460	TYPE,	3-AXIS	1.000	1.000	32.6	54.9	54.9	33.	165.	0.	198.
PROPULSION	0	0	TOT.IMP.	0.	1.000	1.000	0.0	0.0	0.0	0.	0.	0.	0.
SPACECRAFT	13463	12983					55.9	72.2	72.2	57.	217.	0.	274.
MISSION EQUIPMENT	7175	7175	COMPLXTY,	LOW	1.000	1.000	28.3	24.7	24.7	113.	91.	0.	204.
SATELLITE	17633	19758					84.2	96.9	96.9	170.	308.	31.	509.
ASE					1.000		5.9			6.	0.	0.	6.
LAUNCH SUPPORT						1.000		5.8	5.8	0.	0.	23.	23.
GROUND STATIONS										0.	0.	0.	0.
MISCELLANEOUS										1.	5.	0.	6.
SE AND TD										11.	5.	0.	17.
TOTAL										188.	319.	54.	561.

FISCAL YEAR	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
DESIGNS AND REDESIGNS														
SPACECRAFT	1.00													1.0
MISSION EQUIPMENT	1.00					2.00					1.00			4.0
SATELLITE SCHEDULE														
NEW (CURR REUSABLE)	1.	0.	0.	0.	0.	2.	0.	0.	0.	0.	0.	0.	0.	3.
REFURB (RATE=.320)	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	1.	0.	0.	1.

FISCAL YEAR	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
FUNDING																		
RDTE	10.	29.	34.	18.	7.	6.	18.	21.	11.	4.	3.	9.	11.	5.	2.	0.	0.	188.
INVESTMENT	0.	0.	27.	55.	20.	0.	0.	50.	111.	40.	0.	0.	4.	9.	3.	0.	0.	319.
OPERATIONS	0.	0.	0.	3.	3.	0.	0.	0.	6.	6.	0.	0.	0.	18.	18.	0.	0.	54.
TOTAL	10.	29.	61.	76.	30.	6.	18.	71.	128.	50.	3.	9.	15.	32.	23.	0.	0.	561.

Table 7-59. Individual Program Cost Breakdown (1 of 4) - HEAO, 1-yr Life (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES PAYLOADS NEW	REFURB	LAUNCH VEHICLES				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	8.	0.	8.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	23.	0.	23.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	43.	0.	43.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	66.	0.	66.
1979	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	59.	5.	64.
1980	1.	0.	0.0	0.0	0.0	0.0	.7	0.0	0.0	39.	8.	47.
1981	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	48.	5.	53.
1982	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	69.	5.	74.
1983	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	97.	5.	102.
1984	1.	1.	0.0	0.0	0.0	0.0	1.4	0.0	0.0	81.	15.	96.
1985	1.	1.	0.0	0.0	0.0	0.0	1.1	0.0	0.0	53.	12.	65.
1986	0.	2.	0.0	0.0	0.0	0.0	.7	0.0	0.0	56.	7.	63.
1987	0.	2.	0.0	0.0	0.0	0.0	.9	0.0	0.0	46.	9.	55.
1988	0.	1.	0.0	0.0	0.0	0.0	.7	0.0	0.0	29.	8.	37.
1989	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0	24.	4.	28.
1990	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	10.	5.	15.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	4.	12.	0.0	0.0	0.0	0.0	8.4	0.0	0.0	751.	88.	839.

Table 7-59. Individual Program Cost Breakdown (2 of 4) - HEAO, 2-yr Life (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES: PAYLOADS NEW	PAYLOADS REFURB	LAUNCH VEHICLES				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST: PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	8.	0.	8.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	23.	0.	23.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	44.	0.	44.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	54.	0.	54.
1979	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	40.	5.	45.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	51.	0.	51.
1981	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	40.	6.	46.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	41.	0.	41.
1983	0.	1.	0.0	0.0	0.0	0.0	.5	0.0	0.0	62.	5.	67.
1984	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0	59.	4.	63.
1985	1.	0.	0.0	0.0	0.0	0.0	.4	0.0	0.0	39.	4.	43.
1986	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0	37.	4.	41.
1987	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0	40.	5.	45.
1988	0.	1.	0.0	0.0	0.0	0.0	.7	0.0	0.0	39.	7.	46.
1989	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0	16.	4.	20.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	3.	5.	0.0	0.0	0.0	0.0	4.2	0.0	0.0	593.	44.	637.

Table 7-59. Individual Program Cost Breakdown (3 of 4) - HEAO, 4-yr Life (Millions of 1971 Dollars)

LAUNCHED FROM ETR

FISCAL YEAR	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES PAYLOADS NEW	REFURB	LAUNCH VEHICLES				SHTL SHTL	TUG	TUG EXP	PROGRAM DIRECT COST PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	9.	0.	9.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	27.	0.	27.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	56.	0.	56.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	69.	0.	69.
1979	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	30.	5.	35.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	12.	0.	12.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	43.	0.	43.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	92.	0.	92.
1983	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	83.	6.	89.
1984	1.	0.	0.0	0.0	0.0	0.0	.4	0.0	0.0	26.	4.	30.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	14.	0.	14.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	43.	0.	43.
1987	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0	53.	4.	57.
1988	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0	20.	4.	24.
1989	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	3.	2.	0.0	0.0	0.0	0.0	2.2	0.0	0.0	577.	23.	600.

Table 7-59. Individual Program Cost Breakdown (4 of 4) - HEAO, 5-yr Life (Millions of 1971 Dollars)

LAUNCHED FROM ETR

	SPACE TRANSPORTATION SYSTEM SCHEDULED QUANTITIES		LAUNCH VEHICLES							PROGRAM DIRECT COST		
FISCAL YEAR	PAYLOADS NEW	REFURB					SHTL SHTL	TUG	TUG EXP	PAYLOADS	LAUNCH VEHICLES	TOTAL
1975	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	10.	0.	10.
1976	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	29.	0.	29.
1977	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	61.	0.	61.
1978	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	75.	0.	76.
1979	1.	0.	0.0	0.0	0.0	0.0	.5	0.0	0.0	30.	5.	35.
1980	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	6.	0.	6.
1981	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	18.	0.	18.
1982	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	71.	0.	71.
1983	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	128.	0.	128.
1984	2.	0.	0.0	0.0	0.0	0.0	1.4	0.0	0.0	50.	15.	65.
1985	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	3.	0.	3.
1986	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	9.	0.	9.
1987	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	15.	0.	15.
1988	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	32.	0.	32.
1989	0.	1.	0.0	0.0	0.0	0.0	.4	0.0	0.0	23.	4.	27.
1990	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1991	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1992	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1993	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1994	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1995	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1996	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
1997	0.	0.	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.	0.
TOTAL	3.	1.	0.0	0.0	0.0	0.0	2.3	0.0	0.0	561.	24.	585.

C-5

8. Discussion

8. DISCUSSION

8.1 LOW COST DESIGN CONCEPTS

Costs were estimated for each conceptual design at unit and program level. The program level costing was based on the appropriate program traffic from the 1972 NASA mission model. The unit costs were based on the design information developed in the conceptual study effort described in Sections 4, 5, and 6.

Four alternate programs, each having approximately the same scientific value, were compared. The four alternate 12-year programs featured:

(1) 1.5-m (4.92-ft) photoheliograph instrument package flown on seven-day missions

(2) 1.5-m (4.92-ft) photoheliograph instrument package flown initially on seven-day missions then phased into a free flyer

(3) 1.5-m (4.92-ft) photoheliograph instrument package on a free flyer from the start

(4) 1.0-m (3.28-ft) photoheliograph instrument package flown initially on seven-day missions then phased into a free flyer with a 1.5-m (4.92-ft) photoheliograph instrument package

The seven-day missions are one type of Shuttle sortie operation. Comparing the sortie, free flyer, and combination sortie and free flyer program costs, the program with one R&D resulted in the lower overall cost, i.e., one sortie or one free flyer RDT&E for the program duration.

For the sortie/solar observatory the cost of the payload including one sortie mission was determined to be correlative to the mission requirements. By reducing the aperture size from 1.5 m (4.92 ft) to 1.0 m (3.28 ft), the RDT&E plus unit cost was reduced by 50 percent, and by relaxing the pointing accuracy, the RDT&E plus unit cost was reduced by 10 percent.

The automated spacecraft study, using a communications satellite (Intelsat IV) as an example, examined the costs of minor modifications to adapt an existing payload to the Shuttle and for more ideal Shuttle payload configurations. This investigation indicated that the optimum configuration has lower costs for OA demonstration programs such as TDRS with equal or increasing user demands. The minimum spacecraft modification is lowest cost for "one-time" system demonstration programs. If these "one-time" system demonstration programs are for an eventual operational user requiring high availability over long program duration, however, the optimum spacecraft configuration would be lowest cost. If the system demonstration program is a continuing program whereby the payload can be retrived and the spacecraft reused, the optimum spacecraft configuration would also be lowest cost.

The observatory spacecraft study investigated the influence of ground refurbishment and on-orbit service, and design life and revisit schedule on HEAO-C and HEAO program cost. The overall program costs (1979-1990) indicated that on-orbit service is more economical. For early year funding level requirements indicative of the HEAO-C program (1979-1981), however, the ground refurbishment mode has lower cost. The influence of design life and adjustment access schedule were also found to be significant cost drivers. The lowest cost program matching the mission model launch traffic was an HEAO satellite with a two-year lifetime.

8.2 DESIGN GUIDELINES

The design guidelines were developed from the conceptual studies of sortie, automated spacecraft, and observatory spacecraft; reviews of the LMSC design guidelines (Ref. 8.1); and The Aerospace Corporation's experinece in satellite design. These guidelines are all included in Section 5 Volume I, "Reusable Payload Specification." The guidelines which can be traced to this volume are referenced to the appropriate section in this volume.

The conceptual studies provided the major data source for systems analysis and also helped the participants develop guidelines. The study was based on information on three payloads and Shuttle/Tug. The input information on the payload and Shuttle/Tug were of a generally descriptive type and were not of a detail drawing and specification level of input data. The study output (guidelines) is consistent with the input information.

The LMSC design guidelines were reviewed by technical specialists and were rewritten to include those that were considered to be appropriate. Those that were eliminated were considered to have little impact or to be items normally considered as part of a design effort. These rewritten versions along with the Aerospace-developed guidelines are integrated in Section 5 Volume I.

From this study effort it was observed that payload benefits from the Shuttle depend on indoctrinating the designers and subsystem specialists involved in the study. The Shuttle introduces unique features that are not available with current expendable launch vehicles. The current approach of expendable payload design has for over a decade been improving payload performance. To redirect this emphasis to reusable payload and to reduce costs is a departure from current design practice. As more emphasis is placed on Shuttle payloads, more design guidelines will result, as is the case in any design maturity.

8.3 WEIGHT AND PERFORMANCE

For the payloads studied, the gross weight did not exceed the orbiter performance capability, as shown in Figure 8-1. The HEAO design can be forward-mounted, but the longitudinal center of gravity location is marginal. With the payload sharing the aft cargo bay, however, the center of gravity margin will improve. Nevertheless, on retrieval missions there is little assurance that additional payloads will be retrieved to share the bay.

The Intelsat/Tug combination is within the performance and orbiter center of gravity limits if the aft mount is used and the gross weight is limited to the Tug and orbiter performance capability. The maximum ascent orbiter performance is 29,478 kg (65,000 lb) and the maximum Tug performance in the service mode is 1,361 kg (3,000 lb) to geosynchronous orbit. During the orbiter return mode following the Tug retrieval the Tug is basically empty, so the empty gross weight is well within the center of gravity for landing performance limits. For the high payload weights and low performances such as low earth sun synchronous missions, the Tug can be off-loaded because of the low velocity requirement and therefore should be within the performance limits.

The sortie/large solar observatory (LSO) gross weights were within the Shuttle performance capability; but the orbiter center of gravity limits required an aft mounting for the higher gross weight LSO. The lower gross weight concepts can be forward-mounted, as seen in Figure 8-1. The center of gravity was estimated without payload bay sharing, since sharing may not be possible for both deployment and retrieval modes. The sortie/austere solar observatory (ASO) is low in weight $< 6,803$ kg ($< 15,000$ lb) and not restricted to the forward location in the cargo bay. The ASO is small and would share the bay with other sortie payloads.

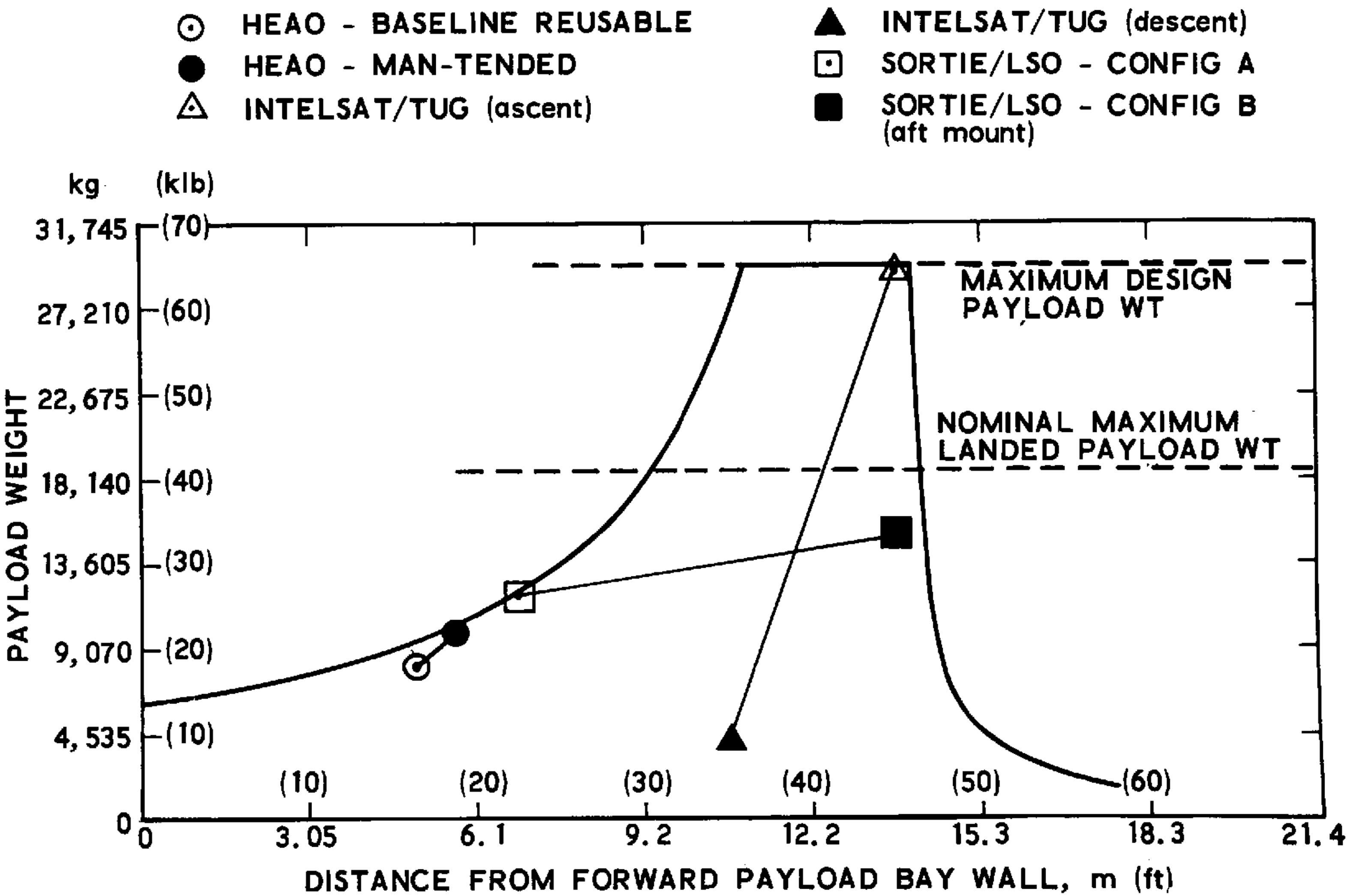

Figure 8-1. Payload Longitudinal Center of Gravity Limits

8.4 REFERENCES

8.1 Design Guide for Low Cost Standardized Payloads, Vol I, LMSC (30 April 1972)